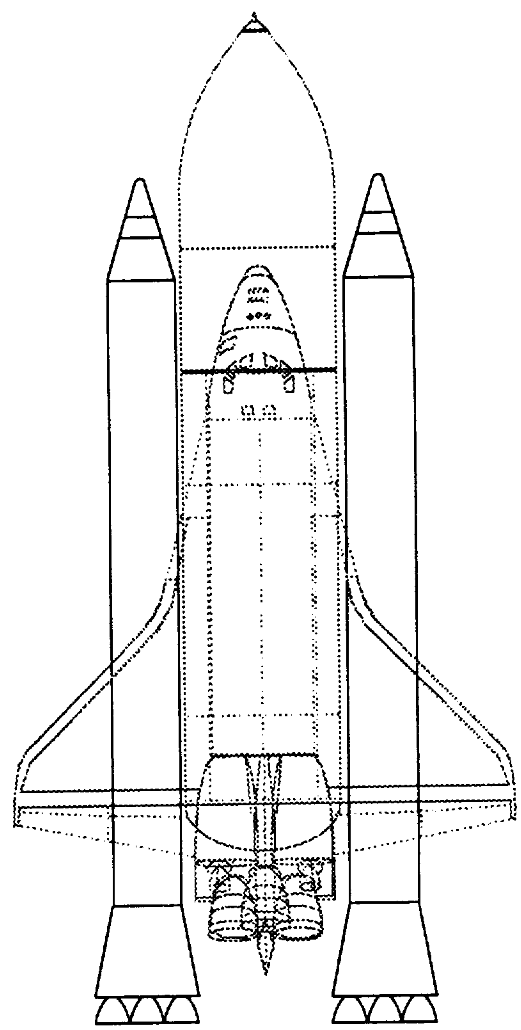
NAS8-37136

March 1989

Appendix G
LRB for the STS
System Study
Level II
Requirements,
Revision 1
January 1988

Liquid Rocket Booster (LRB) for the Space Transportation System (STS) Systems Study



(NASA-CR-183793-App-G) LIQUID ROCKET BOOSTER (LRB) FOR THE SPACE TRANSPORTATION SYSTEM (STS) SYSTEMS STUDY. APPENDIX G: LRB FOR THE STS SYSTEM STUDY LEVEL 2 REQUIREMENTS, REVISION 1 (Martin Marietta   N90-28607

Unclas
G3/20 0251598

MARTIN MARIETTA
MANNED SPACE SYSTEMS

L-004/jer

LRB for the STS System Study
Level II Requirements, Revision 1
January 1988

Appendix G

LIQUID ROCKET BOOSTER
(LRB)
FOR THE SPACE TRANSPORTATION SYSTEM (STS)
SYSTEMS STUDY

LEVEL II REQUIREMENTS
REVISION 1
LRB DOCUM. NO. - TBS
JANUARY 1988

MARTIN MARIETTA
MANNED SPACE SYSTEMS

3.0 REQUIREMENTS.

3.1 SHUTTLE SYSTEM DEFINITION.

3.1.1 Shuttle System Elements.

3.1.1.1 Flight Vehicle Elements. The elements of the Shuttle Flight Vehicle shown in Figure 3.1.1.1 shall be:

a. Orbiter Vehicle

b. Liquid Rocket Booster *

c. External Tank

d. Space Shuttle Main Engine

Characteristics of these elements are defined in Paragraphs 3.3.1, 3.3.2, 3.3.3, and 3.3.4, respectively. The Shuttle Flight Vehicle shall consist of a Shuttle Vehicle Booster, one External Tank, and one Orbiter Vehicle with three Space Shuttle Main Engines.

3.1.1.2 Ground Operations Systems. The major elements of the Shuttle Ground Operations System and the characteristics of these elements are defined in Section 3.4.

3.1.2 Top Level Schematic Block Diagram. The top level schematic block diagram shown in Figure 3.1.2 identifies the Shuttle System elements and other systems with which the Shuttle System interfaces.

3.1.3 Shuttle System Weight and Performance Control. (TBD) *

3.1.4 Integrated Vehicle Configuration. (TBD) *

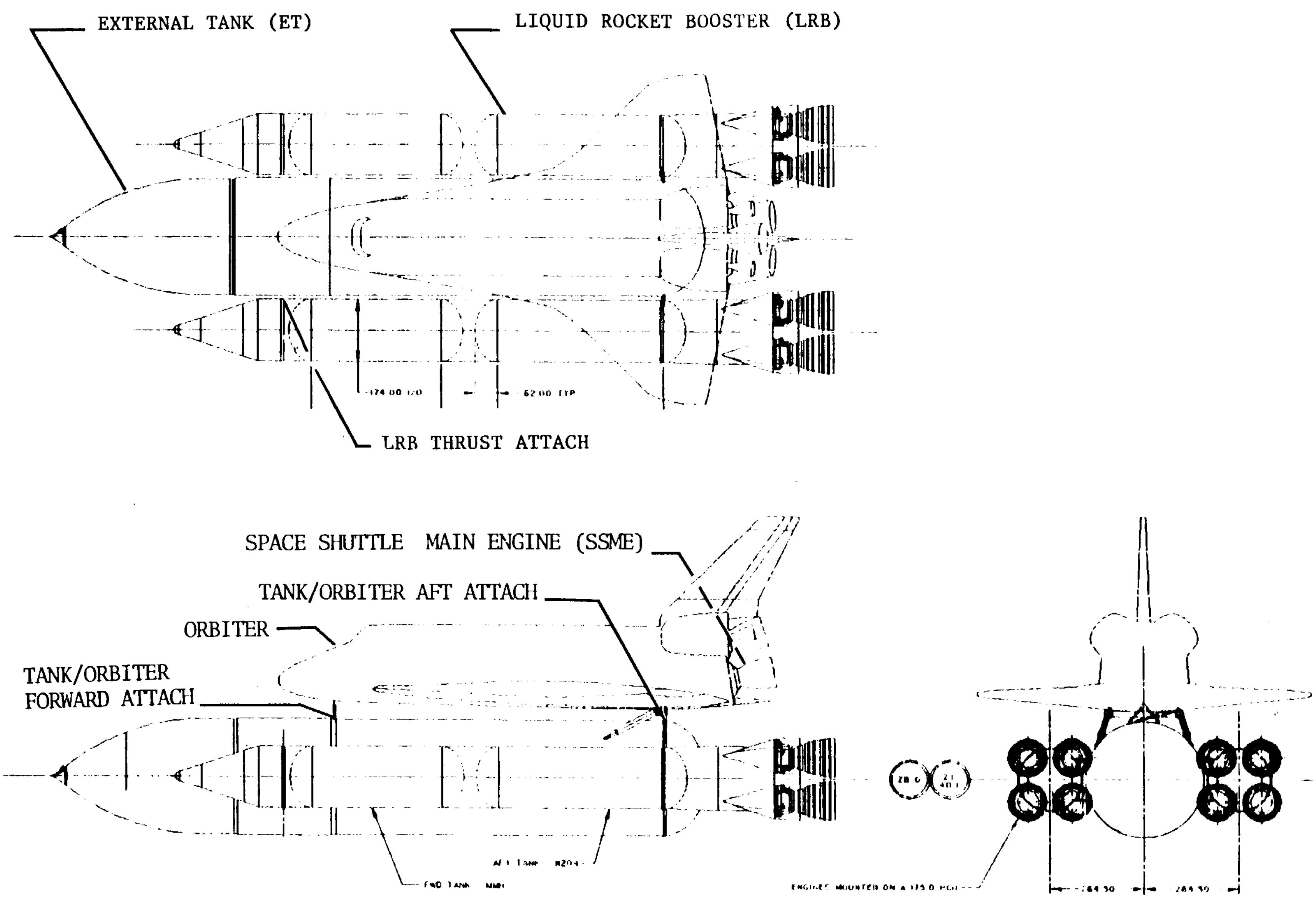


Figure 3.1.1.1 Shuttle Flight Vehicle (General Arrangement, not to Scale-Reference Paragraph 3.1.4 Requirements)

ORIGINAL PAGE IS OF POOR QUALITY

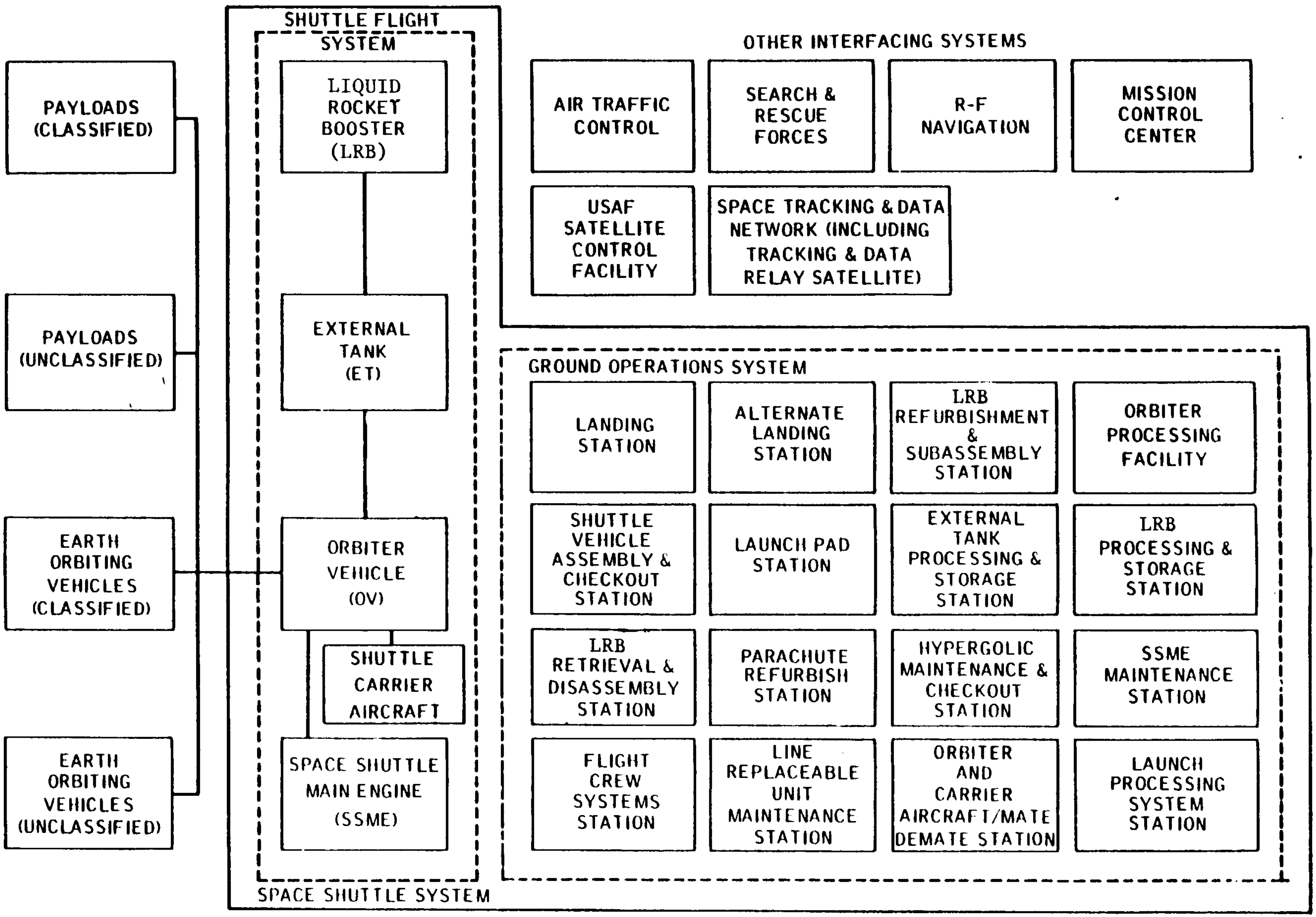


Figure 3.1.2 System Schematic Block Diagram

3.2 PERFORMANCE AND DESIGN CHARACTERISTICS.

3.2.1 <u>Mission Performance</u>. The following subparagraphs specify the performance requirements categorized by the top level functions. *

3.2.1.1 Mission Operations Functions. *

3.2.1.1.1 (Deleted).

3.2.1.1.2 Payload Range. The Space Shuttle Flight Vehicle shall be capable of nominally operating within the payload range from zero to 65,000 lbs. for launch and zero to 32,000 lbs. for entry and landing.

3.2.1.1.2.1 Weight and Volume Chargeable to Payload. All Orbiter scar weight for removable, replaceable items shall be charged to the Orbiter.

3.2.1.1.2.1.1 RCS Propellant. The weight of the RCS consumables required to achieve the pointing accuracy requirements defined in 3.3.1.1.1 shall be chargeable to the payload. RCS consumables control weight shall be (TBD) *

3.2.1.1.2.1.2 Payload Spinup. Spinup capability, if required by the payload, shall be provided by the payload.

3.2.1.1.2.1.3 EVA/IVA Operations. Equipment, expendables, and accessories to support EVA/IVA operations in excess of those specified in 3.3.1.2.4.6 shall be provided at the expense of payload weight and volume. The volume and restraints for the extravehicular mobility unit to support two men EVA/IVA operations shall be provided by the Orbiter. Manned maneuvering units and accessories provided solely for support of payload operations shall be weight and volume chargeable to the payload.

3.2.1.1.2.1.4 Payload Bay Service Panels. The weight difference between standard service panels (with connectors for services as specified in 3.3.1.3.3.2 and 3.3.1.3.3.5.4) and peculiar service panels for payload bay GSE servicing shall be charged against the payload weight.

3.2.1.1.2.1.5 Docking Module. The weight and volume of the docking module shall be charged against the payload as specified in 3.3.1.2.1.3. The docking module control weight shall be as specified in NSTS 07700, Volume X, Appendix 10.12. The control envelope is specified in NSTS 07700, Volume XIV. *

3.2.1.1.2.1.6 Payload Module Atmospheric Control Provisions. Expendables, hardware, and related storage facilities required to accomplish atmospheric control and revitalization for a habitable payload module shall be charged to payload weight and volume as specified in Paragraph 3.3.1.3.3.6.2. The control weight of these provisions shall be in accordance with NSTS 07700, Volume X, Appendix 10.12. The control envelope is specified in NSTS 07700, Volume XIV. *

3.2.1.1.2.1.7 Thermal Control. The weight of thermal control provisions required by a payload which are in excess of that provided by 3.3.1.3.3.6.1 and 3.3.1.3.3.12 shall be charged against the payload weight.

3.2.1.1.2.1.8 OMS Propellant. The weight of OMS consumables in excess of the OMS delta V requirements specified in 3.2.1.1.3 shall be chargeable to payload. The weight and volume of OMS equipment in excess of the storage capacity specified in 3.3.1.2.2.2.2 shall be chargeable to the payload. The OMS delta V kits inert control weight and the control weight of the OMS consumables are (TBD) *

3.2.1.1.2.1.9 Crew Expendables. For missions of more than 28 man-days, the * weight of additional crew expendables as specified in 3.3.1.2.1.1, beyond those required for 28 man-days, shall be charged to the payload. The crew expendables control weight for an additional 14 man-days is specified in NSTS 07700, Volume, X,Appendix 10.12. The volume and hardpoints for the expendables required beyond 42 man-days shall be provided by the payload and located external to the Orbiter cabin.

3.2.1.1.2.1.10 Crew Hardware Provisions. All hardware provisions (over and * above structural hardpoints) for accommodating six additional crewmen in the cabin shall be provided in kit form and the weight charged to the payload (as stated in 3.3.1.2.1.1). The crew hardware provisions control weight is specified in NSTS 07700, Volume X, Appendix 10.12. The volume of hardware provisions required beyond 42 man-days shall be chargeable to the payload and may be located external to the Orbiter cabin.

3.2.1.1.2.1.11 Mission Peculiar Equipment. Special connectors, lines, * cables, monitor and control equipment beyond standard interface provisions as specified in 3.3.1.2.7.3, 3.3.1.2.7.4, 3.3.1.2.7.5, 3.3.1.2.7.6, and 3.3.1.3.3.17 shall be charged to the payload. The control weight for mission peculiar equipment is specified in NSTS 07700, Volume X, Appendix 10.12.

3.2.1.1.2.1.12 Fuel Cell Water Storage. The weight and volume of stored fuel * cell water in excess of the baseline storage capability as specified in 3.3.1.2.4.8 shall be charged to the payload. The equipment control weight for water storage is specified in NSTS 07700, Volume X, Appendix 10.12. The control envelope is specified in NSTS 07700, Volume XIV.

3.2.1.1.2.1.13 Electrical Energy Supply. Provisions for supplying the * electrical energy to a payload in excess of the 50 kWh power provided by the Orbiter as specified in Paragraph 3.3.1.3.3.3.3.4 shall be chargeable to the payload. The control weight of these provisions is specified in NSTS 07700, Volume X, Appendix 10.12.

3.2.1.1.2.1.14 Waste Storage. The weight and volume of condensate and urine storage in excess of the baseline capability as specified in 3.3.1.2.4.5 shall be charged to the payload.

3.2.1.1.2.1.15 Payload Bay Tilt Tables/Swingout Systems. The weight and volume of tilt tables and swingout systems required for payloads as specified in * 3.3.1.2.1.4 shall be charged to the payload. The control weight of these provisions shall be specified in NSTS 07700, Volume X, Appendix 10.12 and the control volume shall be specified in NSTS 07700, Volume XIV.

3.2.1.1.2.1.16 Second Manipulator Arm. The weight of the second manipulator arm shall be charged to the payload as specified in Paragraph 3.3.1.2.1.4. The control weight shall be 905 pounds for the arm and 397 pounds for installation hardware.

3.2.1.1.2.1.17 Encryption/Decryption Equipment. The weight of removable encryption/decryption equipment shall be charged to the payload. The control weight is specified in NSTS 07700, Volume X, Appendix 10.12. *

3.2.1.1.2.1.18 Tunnel Adapter. The weight of the removable tunnel adapter (Reference Paragraph 3.3.1.2.1.9) shall be charged to the payload. The control weight shall be 900 lbs. Any weight for the tunnel adapter in excess of 900 lbs. shall be chargeable to the Orbiter.

3.2.1.1.3 Reference Missions. The missions capability envelope for the Space Shuttle Vehicle is determined by the total requirements and mission functions specified in this document. The reference missions defined herein are within the total capability envelope and shall be used to guide the further definition of functional requirements. The reference missions define typical operational functions required of the Shuttle vehicle and illustrate current operational techniques and philosophy. They serve as an operations baseline against which the vehicle design can be measured. They provide mission data for use in deriving vehicle design requirements and operating environments. Missions 1, 2, and 3 are design reference missions. Mission 4 is a performance reference mission based on the design reference missions.

The Space Shuttle Reference Missions are described below. For performance comparison, Missions 1 and 2 will be launched from Kennedy Space Center (KSC), and Missions 3 and 4 will be launched from the Western Test Range (WTR). The boost phase, which ends Post OMS insertion burn in a standard trajectory and MECO burnout for a direct insertion trajectory, shall provide insertion into an orbit with a minimum apogee of 100 NM, as measured above the earth's mean equatorial radius as defined in NSTS 07700, Appendix 10.10, Section 9.0. The Orbiter SSME cutoff shall be targeted for External Tank disposal. The OMS shall provide the impulse to achieve the desired reference orbit. The mission on-orbit translational delta V capability (in excess of that required to achieve the insertion orbit and that required for on orbit and entry attitude control) is stated for each mission. The Reaction Control Subsystem (RCS) translation delta V required for each mission shall be used to accomplish all post OMS burn rendezvous and docking maneuvers. *

3.2.1.1.3.1 Mission 1. Mission 1 is a payload delivery mission to a 150 NM circular orbit. The mission will be launched due east and requires a payload capability of 65,000 lb. The purpose of this mission is either the placement in orbit of a 65,000 lb satellite or the placement in orbit of a 65,000 lb satellite and retrieval from orbit of a 32,000 lb satellite. The Orbiter vehicle on-orbit translational delta V requirements in excess of 50 x 100 NM reference orbit are 650 ft/sec from the OMS and 100 ft/sec from the RCS.

3.2.1.1.3.2 Mission 2. Mission 2 is a 7 day combination revisit to an orbiting element and spacelab mission. The orbiting element is a 270 NM circular orbit at 55° inclination. The Orbiter vehicle on-orbit translational delta V requirements in excess of 50 x 100 NM reference orbit are 1,250 fps from the OMS and 120 fps from the RCS. The payload capability will be based on existing performance requirements as defined for Missions 1, 3a and 3b.

3.2.1.1.3.3 Mission 3. Mission 3 shall consist of two missions, one for payload delivery and one for payload retrieval. This is a three-day, two-man mission. Mission 3 shall be used only for ascent and entry performance.

3.2.1.1.3.3.1 Mission 3(a). This mission is a payload delivery mission to an orbit of 104° inclination and return to the launch site. The boost phase shall provide insertion into an orbit with a minimum apogee of 100 NM, as measured above the earth's equatorial radius. The Orbiter vehicle on-orbit translation delta V requirements in excess of a 50 x 100 NM reference orbit are 250 fps from the OMS and 100 fps from the RCS. The ascent payload requirement is 32,000 lb. For mission performance and consumables analysis, a return payload of 2,500 lb will be assumed (the 2,500 lb is included in the 32,000 lb ascent payload weight).

3.2.1.1.3.3.2 Mission 3(b). This mission is a payload retrieval mission from a 100 NM circular orbit at 104° inclination. The return payload weight is 25,000 lb. For mission performance and consumables analysis, an ascent payload of 2,500 lb will be assumed (the 2,500 lb is included in the 25,000 lb return payload weight). The Orbiter vehicle on-orbit translation delta V requirement in excess of 100 NM circular orbit is 425 ft/sec from the OMS. The translation delta V requirement from the RCS is 190 ft/sec.

3.2.1.1.3.4 Mission 4. This is a Performance Reference Mission. This mission is a payload delivery and retrieval mission launched from Vandenberg Air Force Base launch site into a final inclination of 98° in a 150 NMI circular orbit as measured above the earth's equatorial radius. The ascent cargo weighs 32,000 lb and has a 15 ft dia x 60 ft long envelope. The mission shall deploy a spacecraft weighing 29,500 lb within two revolutions after lift-off. Upon subsequent completion of necessary phasing and rendezvous maneuvers, a similar passive-cooperative stabilized spacecraft weighing 22,500 lb shall be retrieved from a 150 NMI orbit and returned to VAFB. The mission duration shall be 7 days for mission performance and consumables analysis. A spacecraft cradle weight of 2,500 lb must be added to the return spacecraft weight. The RCS will be loaded full at lift-off, and the minimum translation delta V from the OMS, including post meco-insertion burn, is a total of 1,050 fps. Standard provisions shall be included for personnel and stowed equipment, and contingency EVA capability shall be provided.

3.2.1.1.4 Ascent Performance. The flight vehicle ascent performance and payload capability shall be based on nominal ISPs for OMS, RCS, and the main engine delivered specific impulse value as stated in the SSME/Orbiter Vehicle ICD 13M15000, nominal value for a single engine. Performance of the LRBs shall be as specified in Paragraph 3.3.2.1.2. For design missions, SSME power level at lift-off shall not exceed 100 percent. Power levels above 100 percent will be attained following the high-q throttle-down period. The exception to this rule would be in the case of loss of power of one or more SSMEs or LRB engines, in which case the remaining SSME(s) may be throttled to 109 percent. Flight performance reserves shall be based on ± 3 sigma systems and environment dispersions, except during the AOA/ATO abort portion of the missions. The flight performance reserves during the AOA/ATO portion of the missions shall be based on ± 2 sigma systems and environmental dispersions. *

3.2.1.1.4.1 Yaw Steering. The flight vehicle shall have the capability of yaw steering, first stage flight, to accommodate aerodynamic sideslip angle control to a nominal value of zero in the transonic flight region for a smoothed design wind condition. yaw steering during second stage flight shall be provided to afford operational flexibility in accommodating communications constraints, ET disposal constraints, and intact abort to a high crossrange target.

3.2.1.1.5 Propellant Dump. During ascent, after SSME shutdown and ET separation, the Orbiter shall be capable of dumping propellants remaining trapped in the MPS feedlines and main engine. Dumping through the main engine will be initiated during OMS burn to insertion, or for direct insertion trajectory, at the same time a normal insertion burn would have taken place.

3.2.1.1.6 Insertion Accuracy. The guidance and control subsystem in conjunction with the autonomous onboard navigation subsystem shall produce an Orbiter state vector at MECO with one sigma dispersons relative to the desired state vector no greater than shown in Table 3.2.1.1.6, for the following conditions:

a. 3 IMU operation - No failures

b. Total time between IMU align complete discrete being set and launch not to exceed 33 minutes

c. No SSME failures within 30 seconds of MECO

d. MECO conditions defined as SSME chamber pressure less than 1% on all engines

3.2.1.1.7 Day and Night Operations. The Shuttle System shall have the capability to launch and land the flight vehicle in daylight or darkness. The Orbiter shall be capable of terminal rendezvous and retrieval of a cooperative target, under daylight and darkness conditions. The Shuttle System shall be capable of support EVA operations under daylight and darkness conditions.

3.2.1.1.8 DOD Missions. The Shuttle Flight Vehicle shall be capable of performing the DOD missions independent of ground support from ground stations outside the contiguous U.S. for normal operations. It shall contain provisions for the installation of GFE COMSEC equipment for encryption/decryption/ authentication for classified operations.

This does not preclude use of the AFSCF for secure voice transmission for support in the event of an emergency, nor dues it preclude use of navigational/ communications satellites which simultaneously service multiple users independent of Shuttle operations, nor does it restrict use of ground base terminal landing aids, nor does it preclude the use of launch site tracking.

3.2.1.1.9 Shuttle Vehicle Separation - Nominal Modes. The separation * subsystem(s) shall provide for Shuttle element separation without damage to or recontact of the elements during or after separation. Damage to the LRB/ET connectors on the aft upper struts at the LRB/ET interface during LRB separation after the ATVC power is deadfaced is acceptable. Nominal modes shall include conditions resulting from trajectories which include dispersions but which preclude failures specified in 3.2.1.5. The nominal separation modes are:

a. Separation of the LRBs from the Orbiter/ET at staging

b. Separation of the ET from the Orbiter after Main Engine Cutoff (MECO).

3.2.1.1.9.1 LRB Separation. Separation of the LRBs from the Orbiter/ET shall * occur only after LRB shutdown. The separation shall be automatically inhibited if vehicle body rates and/or dynamic pressure exceed those values for which the separation system has the capability to perform a separation without causing damage to or recontact of Shuttle elements, with the exception of damage to the aft LRB/ET electrical connectors after ATVC power is deadfaced. The crew shall be provided the capability to manually override these body rate and dynamic pressure inhibits.

3.2.1.1.9.1.1 LRB Separation. The LRB separation system shall include: *

a. Separation flight control functions

b. Release system

c. Booster Separation Motor (BSM) system

The LRB separation system shall incorporate signal interlocks to prevent LRB * release and BSM ignition due to stray signals. The separation system shall not release any debris which could cause damage to any Orbiter/ET system or subsystem during separation under conditions specified in Paragraph 3.2.1.1.9.1.3, Design LRB Staging Conditions. *

3.2.1.1.9.1.1.1 Separation Flight Control Functions. Separation flight control functions consist of flight control system functions necessary to support the separation sequence specified in Paragraph 3.2.1.1.9.1.2. These shall include:

a. Return of the nozzles of each LRB to a position 0.0 ±1.0 degree from the LRB centerline in the vehicle pitch axis * and 1.0 ±0.6 degrees from the LRB centerline, toward the External Tank, in the vehicle yaw axis. This position shall be maintained for at least 5 seconds after separation command issuance.

b. Transition of the flight control system configuration from that for Orbiter/ET/LRB flight to that for Orbiter/ET flight.

c. Separation-required control of vehicle attitude and/or attitude rate.

3.2.1.1.9.1.1.2 Release System. The release system shall be compatible with * the separation sequence specified in 3.2.1.1.9.1.2. Any component disconnect or breakwire at release shall not induce an impulse torque in excess of 700 ft-lb-sec about the LRB CG at separation.

3.2.1.1.9.1.1.3 Booster Separation Motor System. Separation motors shall be * installed in a forward LRB position (nose cone frustum) and in an aft position (aft skirt). At both the forward and aft locations there shall be a cluster of four BSMs. At both locations, the thrust vector of the BSM cluster shall be parallel to ±4 degrees to a plane containing the LRB centerline which is rotated 20 degrees about the centerline from the LRB +Z axis toward the ET (Figure 3.2.1.1.9.1.1.3). The thrust vector of the forward cluster shall pass within 2.6 inches of the LRB centerline. The thrust vector of the aft cluster * shall be offset 1.95 ±3.9 inches from the LRB centerline toward the ET in a direction normal to the 20 degree plane. In addition, the thrust vector of each cluster shall be pitched, in the 20 degree plane, 40 ±4 degrees from the LRB Y-Z plane; the forward cluster shall be pitched forward and the aft cluster shall be pitched aft.

The BSMs shall be designed to operate over a propellant bulk temperature range of 30 degrees F to 120 degrees F. Each cluster of four motors shall provide the following vacuum performance over the entire propellant operating temperature range.

a. Average thrust over the web action time _ (TBD) lbs. *

b. Neutral or regressive chamber pressure trace

c. Total impulse over the web action time _ (TBD) lb-sec. *

d. Total impulse over the action time _ (TBD) lb-sec. *

e. Thrust rise characteristics compatible with sequencing requirements specified in 3.2.1.1.9.1.2

f. The time from BSM ignition start until the chamber pressure during thrust tail-off is one-half the chamber Pressure at End of Web Action Time (PEWAT/2) shall not exceed 1050 milliseconds for each BSM.

g. Web action time _ 0.8 seconds for each BSM.

The BSMs shall not release any debris which could damage the Orbiter TPS during separation under conditions specified in Paragraph 3.2.1.1.9.1.3, Design LRB Staging Conditions. The BSM-induced Orbiter/ET thermal environment is shown in NSTS 07700, Volume X, Appendix 10.11.

3.2.1.1.9.1.2 LRB Separation Sequence. Initiation and control of the LRB separation sequence shall be the responsibility of the Orbiter. The primary * cue for initiation of the separation sequence shall be (TBD). The backup cue shall be mission elapsed time.

Each LRB shall furnish redundant (TBD) signals to the Orbiter during LRB thrust shutdown.

The following commands shall be issued at a time from sequence initiation which assures that both LRB nozzles are positioned as specified in Paragraph 3.2.1.1.9.1.1.1 at the time of separation command:

a. Null LRB Thrust Vector Control (TVC) actuators *

b. Initiate Orbiter/ET flight control center configuration

Separation-required control of vehicle attitude and/or attitude rate shall be initiated at a time from sequence initiation which assures its effective operation. It shall be terminated no sooner than 4.0 seconds after separation command issuance.

The LRB separation command shall be issued at a time from sequence initiation which assures that the thrust of LRB is less than or equal to (TBD) pounds. *

The LRB separation system shall provide for concurrent initiation of the release and BSM ignition of both LRBs. Release of all structural attachments shall occur within 30 milliseconds and the vacuum thrust of each cluster of four BSMs shall reach TBD pounds within 30 to 135 milliseconds of the time at which the separation command crosses the Orbiter/LRB interface.

LRB residuals venting after separation (TBD) *

3.2.1.1.9.1.3 Design LRB Staging Conditions. The LRB separation system shall be designed to provide a safe separation for staging conditions which comprise any combination of values, within the specified limits, of these parameters: *

a. Roll rate between -5^{o}/sec and $+5^{o}$/sec

b. Pitch rate between -2^{o}/sec and $+2^{o}$/sec

c. Yaw rate between -2^{o}/sec and $+2^{o}$/sec

d. Dynamic pressure less than or equal to 75 psf

The separation system shall be designed to provide a safe separation for pitch and sideslip angles at staging which do not exceed ± 15 degrees.

3.2.1.1.9.2 Orbiter/ET Separation. Orbiter/ET separation shall include:

a. Fluid line and electrical umbilical disconnect

b. Retraction of Orbiter umbilicals

c. Structural attachment release

d. Maneuvering of the Orbiter away from the ET

Performance and sequencing of these functions shall be initiated and controlled by the Orbiter vehicle. The release hardware shall be the responsibility of the Orbiter.

3.2.1.1.9.2.1 Orbiter/ET Separation Performance. The Orbiter/ET separation subsystem shall provide safe separation for the conditions specified in Paragraph 3.2.1.1.9.2.3. The separation structural release shall be automatically inhibited if a propellant feed umbilical disconnect valve fails to close or if the body rates exceed those values for which the separation system has the capability to perform a separation without causing damage to or recontact of Shuttle elements. The ability to manually inhibit and subsequently enable release and bypass an automatic structural release inhibit shall be provided. The operation of the separation subsystem shall not result in the release of any debris.

The RCS shall provide a delta V _ 4 fps along the -Z axis to the Orbiter for separation. This shall be accomplished using the forward and aft RCS to provide the maximum -Z axis acceleration consistent with insertion attitude control requirements.

3.2.1.1.9.2.1.1 Separation Flight Control Requirements. Separation flight control functions consist of the FCS functions necessary to support the sequences specified in 3.2.1.1.9.2.2. These shall include:

a. Rate control of the mated Orbiter/ET from separation sequence initiation to structural release within the limits specified in 3.2.1.1.9.2.3.

b. Attitude control during the translation maneuver specified in 3.2.1.1.9.2.1.

3.2.1.1.9.2.2 Orbiter/ET Separation Sequence. The Orbiter/ET separation sequence is initiated when MECO initiation, automatic or manual, is verified. Following this time, time sequenced commands are issued to arm all separation subsystem PICs for closure of LH_2/LO_2 disconnect valves, Orbiter/ET electrical deadfacing, umbilical release and retract, and firing of the structural release pyrotechnics. The ET tumble valve system is also armed after MECO. Firing of the RCS-Z jets is initiated 160 ms prior to structural release. Automatic attitude control will be inhibited until sufficient VZ is available to ensure separation margins. (Note: manual override and manual attitude control are available at any time after structural release except during the automatic attitude control inhibit phase.) The RCS shall then continue with a high mode - Z axis - attitude hold translation maneuver as specified in 3.2.1.1.9.2.1. The separation sequence is terminated after all separation controlled functions have been completed.

Release of all structural attach points shall occur within 0.020 seconds. The automatic separation sequence shall incorporate automatic inhibit of structural release as specified in 3.2.1.1.9.2.1. Automatic structural release inhibits due to excessive body rates are maintained until the body rates fall within acceptable limits or until manual override of the inhibits is initiated. Automatic inhibits of structural release due to disconnect valve failure must be manually overridden after a procedural delay to allow ET pressure relief.

Manual inhibit of separation shall inhibit all separation functions unless these functions have been commanded prior to initiation of the manual inhibit.

3.2.1.1.9.2.3 Orbiter/ET Design Staging Conditions. The Orbiter/ET separaion system shall be designed to provide a safe separation for staging conditions which comprise any combination of values within the specified limits of the following variables:

a. Pitch rate between $-.7^{o}$/sec and $+0.7^{o}$/sec

b. Roll rate between -0.7°/sec and +0.7°/sec

c. Yaw rate between -0.7°/sec and +0.7°/sec

3.2.1.1.10 Shuttle Vehicle Separation-Abort Modes. The separation subsystem(s) shall provide for safe separation under intact abort conditions specified in3.2.1.5.1 The related separation modes shall be: (a) LRB separation from the Orbiter/ET at shutdown under conditions resulting from any of the failures specified in 3.2.1.5.1.3; (b) Orbiter/ET separation under conditions corresponding to SSME cutoff for an Abort-Once-Around (AOA); (c) Orbiter/ET separation at SSME cutoff for conditions corresponding to a Return to Launch Site (RTLS) abort; and (d) Orbiter/ET separation at SSME cutoff for conditions corresponding to TAL abort. *

3.2.1.1.10.1 Abort LRB Separation. Separation of the LRBs from the Orbiter/ET in the event of an abort shall occur only after LRB shutdown. The separation shall be automatically inhibited if vehicle body rates and dynamic pressure exceed those values for which the separation system has the capability to perform a safe separation (contact between LRBs after separation and degradation of Orbiter TPS lifetime by BSM exhaust impingement are acceptable). The crew shall be provided the capability to manually override these body rates and dynamic pressure inhibits. If less than three SSMEs are operating at LRB separation, the separation command shall be issued at a time from separation sequence initiation which assures that the thrust of each LRB is less than or equal to (TBD) pounds. * *
With the exceptions noted above, in an abort the LRB separation system shall meet all requirements specified in Paragraphs 3.2.1.1.9.1.1 through 3.2.1.1.9.1.2.

3.2.1.1.10.2 Abort Separation of Orbiter/ET. Abort separation of the Orbiter shall include:

a. Fluid line and electrical umbilical disconnect

b. Retraction of Orbiter umbilicals

c. Structural attachment release

d. Maneuvering of the Orbiter away from the ET

Performance and sequencing of these functions shall be initiated and controlled by the Orbiter vehicle. The release hardware shall be the responsiblity of the Orbiter.

3.2.1.1.10.2.1 Orbiter/ET TAL/ATO/AOA Separation. The Orbiter/ET separation for Trans-oceanic Abort Landing (TAL) Abort-to-Orbit (ATO) and Abort-Once-Around (AOA) shall be as specified in 3.2.1.1.9.2 through 3.2.1.1.9.2.3 except the -Z shall be 11.0 fps for TAL.

3.2.1.1.10.2.2 Orbiter/ET Abort Separation Performance (RTLS). The Orbiter/ET separation subsystem shall provide safe separation for the conditions specified in 3.2.1.1.10.2.4. The separation structural release shall be automatically inhibited if the angle of attack, sideslip angle or body rates exceed those values for which the separation system has the capability to perform a separation without causing damage to or recontact of Shuttle elements.

The ability to manually inhibit and subsequently enable release and to bypass an automatic structural release inhibit shall be provided. In addition, the separation sequence shall provide a time override of automatic inhibits. The operation of the separation subsystem shall not result in the release of any debris.

The Orbiter/ET separation shall be performed with ET usable propellants ranging from zero to a maximum of 2 percent of propellant loaded at lift-off. The separation shall be accomplished using the forward and aft RCS to provide the maximum -Z axis acceleration consistent with attitude control requirements during a timed separation maneuver. The duration of the translation maneuver shall be such that safe separation can be accomplished for the conditions specified in 3.2.1.1.10.2.4.

3.2.1.1.10.2.2.1 Separation Flight Control (RTLS). Separation flight control functions shall consist of the flight control system functions necessary to support the sequence specified in 3.2.1.1.10.2.3. These shall include:

a. Attitude and rate control of the mated Orbiter/ET from separation sequence initiation to structural release within the limits specified in 3.2.1.1.10.2.4.

b. Attitude and rate control of the Orbiter during the -Z translation maneuver as specified in 3.2.1.1.10.2.4.

3.2.1.1.10.2.3 Orbiter/ET Separation Sequence (RTLS). The Orbiter/ET separation sequence is initiated when MECO initiation, automatic or manual, is verified. Following this time, time sequenced commands are issued to arm all separation subsystem PICs, for closure of the LH_2/LO_2 disconnect valves, Orbiter/ET electrical deadfacing umbilical release and retract, and firing of the structural release pyrotechnics. The ET tumble valve system is also armed after MECO. The RCS shall then provide a high mode -Z axis translation maneuver. The separation sequence is terminated after all separation controlled functions have been completed.

Release of all structural attach points shall occur within 0.02 seconds. The translation maneuver shall be initiated no later than 0.05 seconds following issuance of the structural release command. The automatic separation sequence shall incorporate automatic inhibit of structural release as specified in 3.2.1.1.10.2.2. The separation sequence shall incorporate a timed override of automatic inhibits.

3.2.1.1.10.2.4 Orbiter/ET Design Staging Conditions (RTLS). The Orbiter/ET separation subsystem shall be designed to provide safe separation for the range of conditions shown in Figure 3.2.1.1.10.2.4.

3.2.1.1.10.2.5 Orbiter/ET Contingency Abort Separation. A manually initiated fast ET separation sequence shall also be provided in accordance with Paragraph 3.2.1.5.2.3, which will initiate separation in minimum time during first stage flight.

3.2.1.1.11 Flight Personnel Flight Loads. As experienced by the flight personnel, flight vehicle launch trajectory resultant load factors shall not exceed 3 g's and Orbiter vehicle entry trajectory resultant load factors shall not exceed 3 g's. These load factors are static and do not include dynamic effects. These load factor limits do not apply to abort modes. The product of g forces and time shall not be detrimental to the flight personnel.

3.2.1.1.12 Orbiter Vehicle Attitude Constraints. While the payload bay doors are open, the Orbiter shall have the capability to provide heat removal from the payload up to 29,000 Btu/hr. During on-orbit operations, the Orbiter fixed attitude hold time capability depends on a combination of the following: sun angle relative to the orbit plane (beta angle), Orbiter altitude, Orbiter attitude and previous attitude history, Orbiter and payload heat rejection requirements, water management for heat rejection, and thermal conditioning requirements. Depending on the combination of these factors, the Orbiter allowable hold time capability varies from 5 to 160 hours.

Orbiter pre-entry thermal conditioning attitude may require up to 12 hours of duration depending on the thermal state of the Orbiter prior to the pre-entry attitude initiation. Also the Orbiter ATCS radiators will normally be cold soaked for a minimum of 1 hour in tail to the sun attitude or equivalent prior to closing the payload bay doors for entry.

Specific Orbiter vehicle attitude constraints are defined in NSTS 07700, Volume XIV, Attachment 1, ICD 2-19001, paragraph 6.

3.2.1.1.13 On-orbit Rescue Operations. The design shall provide the capability to perform on-orbit rescue operations. If the spacecraft requiring aid has a docking system on that mission, the primary rescue mode will be by docking, with crew transfer through a pressurized tunnel. Otherwise, emergency rescue will be with pressure suits and personal rescue systems outside the spacecraft.

3.2.1.1.14 Orbiter Direct Entry. The Orbiter vehicle shall have the capability for deorbit and direct entry from a (TBD) orbit with 32,000 lbs. return *
payload. The crossrange associated with this direct entry condition is (TBD) nautical miles.

3.2.1.1.15 Post-Landing Thermal Conditioning. The Orbiter thermal control design shall be based on GSE ground thermal conditioning available within 45 minutes after touchdown for vehicle structural cavities and 45 minutes for the Active Thermal Control Subsystem (ATCS). In an emergency condition, the absence of post-entry/landing GSE cooling will not preclude reuse of the Orbiter vehicle. Any hazardous condition (i.e., possible venting OMS/RCS propellants, cabin overtemperatures, etc.) which results from the absence of ground cooling shall be identified.

3.2.1.1.16 Flight Vehicle Launch CG. (TBD) *

SSV CG LOCATION (SHUTTLE COORDS, IN.)

FLIGHT MODE	Xs	Ys	Zs
Lift-off	± 02	± 2	± 02
Pre-LRB Sep	± 20	± 2	± 12
MECO	± 25	± 2	± 15

3.2.1.1.16.1 Lift-Off Clearances. Position clearance shall exist between the Space Shuttle launch vehicles and all ground launch facility hard points from LRB ignition through tower clearance for both nominal and intact abort modes. Vehicle clearance and drift during lift-off shall be within the envelopes specified in ICD2-0A002. *

3.2.1.1.17 ET Disposal. The SSME cutoff targeting shall be selected such that the nominal ET impact will be in a preselected impact area for both ETR and WTR launch, including the reference missions defined in Paragraph 3.2.1.1.3. The ET impact area is driven by the mission's apogee altitude, type of orbit insertion (standard or direct), and footprint size which are all a function of the MECO target. The footprint size is also dependent on the type tank rupture and breakup (violent or benign) upon reentry into the atmosphere. The ET impact footprint shall fall in either the Indian or Pacific Oceans for all ETR launches. For all WTR launches, the ET impact footprint shall fall in either the Pacific, Antarctic, or Indian Oceans. The preselected impact locations, defined by the External Tank footprint, shall adhere to the following constraints:

a. For nominal missions, the ET impact footprint shall be no closer than 200 n. mi. from foreign land masses; 25 nm from U.S. territories and CONUS (only when mission objectives and performance dictate), and 25 n. mi. from the permanent ice pack of Antarctica.

b. For planned guided MECO abort missions, the ET impact footprint shall not impact land masses. For MECO underspeeds, land impacts shall be minimized.

The approved orbit inclination for missions launched from ETR are between 57 deg. N and 28.5 deg. N. The approved orbit inclination for missions launched from WTR are between 68 deg. S and 99 deg. S. For all missions outside the approved inclinations, special Range Safety approval will be required.

3.2.1.1.18 EVA Operations. The Shuttle System shall provide the capability for extravehicular operations by two crewmen for periods of up to six hours outside the vehicle.

The capability shall also be provided during the orbital flight test phase for extravehicular Manned Maneuvering Unit (MMU) operation in the immediate vicinity of the vehicle for the purpose of flyaround inspection and possible inflight repair activities. The manned maneuvering capability shall be available for operational missions when required to support payload operations. The MMU will be stowed in the payload bay for all flights which require it. MMU weight shall be charged to payloads if flown for payload support.

3.2.1.1.19 (Deleted).

3.2.1.1.19.1 The Orbiter will be provided with high and low frequency (37 and 10 kHz) self-contained, water-actuated acoustic beacons on the payload bay DFI pallet in a manner to ensure activation in the event of Orbiter immersion.

3.2.1.1.19.2 The LRBs will be provided with high and low frequency (37 and 10 kHz) self-contained, water-actuated acoustic beacons on the forward skirt upper ring in a manner to ensure activation in the event of LRB immersion. *

3.2.1.2 Assembly and Launch Functions (FFD 2.0). *

3.2.1.2.1 Notification for Launch. To fulfill the space rescue role, the Shuttle System shall be capable of launching within 26.5 hours after notification with the flight vehicle mated and ready for transfer to the pad. This time includes retargeting to a dissimilar mission, loading a validated flight program, and filling the OMS and RCS propellant tanks. *

3.2.1.2.2 Launch from Standby. The Shuttle System shall have the capability to launch the flight vehicle from a standby status within 4 hours. Vehicle access shall be permitted for not less than 45 minutes of consecutive time within the 4 hours to accommodate flight crew ingress and final prelaunch closeout. The Shuttle System shall have the capability to hold in a standby status up to 24 hours. *

3.2.1.2.3 Cryo Loading. The Shuttle System shall be capable of loading ascent cryogenic propellants within the constraints specified in Paragraph 3.2.1.2.2. The design shall not preclude main propellant drain and subsequent reload with no manual operations on the launch pad.

3.2.1.2.3.1 Cryo Loading Monitor and Control. The Shuttle Ground System shall be capable of monitoring and remotely controlling flight vehicle functions and parameters critical to propellant loading or draining.

3.2.1.2.3.2 Hold After Cryo Loading. With due consideration to internal subsystems management, the Shuttle System shall be capable, without recycle, of holding after LRB and MPS propellant loading for at least seven hours prior to the initiation of LO_2 drainback. Subsequent to the initiation of LO_2 drainback, a two minute hold capability, with reduction of vehicle performance capability, shall exist until T-31 seconds. *

3.2.1.2.4 Payload Changeout. The Shuttle System shall be capable of performing on-pad payload changeout as specified in 3.2.1.2.1 and 3.3.1.1.6. The specified environmental contamination control requirements in 3.6.12.2 and DOD control requirements shall be maintained during the exchange of a payload assembly at the launch pad.

3.2.1.2.5 On-Time Launch. From initiation of launch activities (beginning of standby through lift-off or from the beginning of the countdown through lift-off) the Shuttle System shall be capable of achieving a lift-off with $\pm$ two seconds of the target lift-off time GMT. The two second tolarance shall apply to flight vehicle subsystems only. The ground systems functional reliability shall be in accordance with 3.5.1.2.

3.2.1.2.6 Vehicle Launch Orientation. The Shuttle Flight Vehicle shall be in a tail south orientation for launch at KSC; for launches from WTR, the vehicle shall be oriented tail west.

3.2.1.2.7 Propellant Fill (TBD) *

3.2.1.2.7.1 RCS Propellant Fill. The RCS tanks will be loaded full. The ground systems will provide the capability to vacuum (less than 1 psia) fill the RCS manifold.

In addition, the RCS tankage shall also have the capability to be offloaded, using the PVT method, to a minimum 65% (lb wt) of maximum rated loading for * specific selected missions as deemed necessary.

3.2.1.2.7.2 EPS Cryogenic Reactant Fill. The Shuttle System shall be capable of offloading electrical power subsystem cryogenic reactants for specific * selected missions, as deemed necessary.

3.2.1.2.8 Prelaunch Purge. All Shuttle elements shall utilize GSE and facilities to meet all purge requirements during the prelaunch phase.

3.2.1.2.9 On-Pad Abort. The vehicle shall be capable of recycling to the main engine start sequence within 24 hours subsequent to a SSME or dual LRB engine * shutdown prior to liftoff. Subsequent to an on-pad abort, the Shuttle System shall have the capability to accomplish the rescheduled design mission without rollback to the VAB for vehicle TPS refurbishment and/or recertification.

3.2.1.2.9.1 Emergency Power to accomplish Abort. The Shuttle system shall * have the ability to accommodate the full loss of thrust of one LRB engine on each LRB and successfully complete an intact abort. LRB engines may be throttled up to emergency power (100%) to accomplish this.

3.2.1.2.10 Retargeting. The Shuttle shall be capable of retargeting to a dissimilar mission within 16 hours. The design of ground and flight systems * shall not preclude the capability to retarget within 2 hours.

3.2.1.2.11 Pad Stay Time. The Space Shuttle System shall accommodate the mated vehicle on the launch pad for durations up to 180 days. Exposure to natural and induced environments for the pad stay time duration shall not invalidate the design performance or operational capability of the flight vehicle.

3.2.1.2.12 Emergency Egress. Emergency egress shall provided for crew and passenger evacuation to a safe area in a maximum time of 2 minutes (from * crew/passenger ingress up to LRB ignition).

3.2.1.2.13 Cabin Pressure Integrity Verification. The Space Shuttle System shall be capable of pressurizing the crew module up to 2 psid through the cabin hatch and venting the crew module through onboard valves while on the launch pad and after crew ingress and cabin hatch closeout.

3.2.1.2.14 Debris Prevention and Ice Suppression. The Shuttle System, including the ground systems, shall be designed to preclude the shedding of ice and/or other debris from the elements during prelaunch and flight operations that would jeopardize the flight crew and/or mission success.

a. Ice is defined as frozen water of 18 lbs/ft^3 or greater density formed on the outside exposed surface(s) of any element. Frozen water of 18 lbs/ft^3 is considered to be frost and is of no concern.

b. Debris is defined as "broken, scattered remains emanating from the exterior surface(s) of any element".

c. NSTS 16007, Shuttle Launch Commit Criteria and Background Document, contains the specific External Tank locations where the design does not preclude the formation of ice/frost. *

3.2.1.2.14.1 The Shuttle System shall be designed so that "Launch Holds" due to ice formation shall not occur more than 5% of the time based on atmospheric conditions at the launch pad in the proximity of applicable launch vehicle surfaces.

3.2.1.2.14.2 The Shuttle System shall provide the capability to monitor the local atmospheric conditions and provide an ice suppression system if the probability of launch holds due to ice formation exceeds 5% as defined in Paragraph 3.2.1.2.14.1 above. *

a. The ice suppression system shall be designed to maintain the external tank surface temperature at 33 degrees F or above, ET surface temperature not to exceed 130 degrees F, LRB surface temperature (including AFT skirt area) limits - TBD, Orbiter surface temperature not to exceed 100 degrees F (exposure duration not to exceed 7 hours) and SSME engine nozzle temperature not to exceed 100 degrees F. *

b. The launch commit criteria shall be based on no ice on these areas of the external tank or LRB tanks. *

3.2.1.2.15 (Deleted).

3.2.1.2.16 (Deleted).

3.2.1.2.17 Secure Communications. The Space Shuttle System shall be capable of providing communicatons security between the Orbiter and the Launch Control Center and between the Orbiter and the Mission Control Center. For GSTDN communications, this will involve both GPC and flight crew control of the command inhibit function. For TDRSS and SGLS communications, it shall include voice and command data encryption and command authentication on the forward link, and operational telemetry data and voice encryption on the return link. The same techniques are to be used during prelaunch checkout as during flight. In addition, for DOD missions the launch databus must be protected to handle classified data.

3.2.1.2.18 24-Hour Scrub/Turnaround. The Space Shuttle System shall be capable of launching from KSC within 24 hours after scrubbing a launch attempt. Scrub may occur any time prior to H_2 igniter ignition.

3.2.1.3 Turnaround Maintenance Operations Functions. *

3.2.1.3.1 Space Shuttle System. The Space Shuttle System, including the Orbiter vehicle, liquid rocket boosters, external tank, vehicle assembly facilities, and launch complex, shall be capable of supporting the planned launch schedule within the time constraints specified in 3.5.2.1, utilizing programmed turnaround resources. *

3.2.1.4 Mission Operations Support Functions. *

3.2.1.4.1 Natural Environment Data Requirements

3.2.1.4.1.1 Meteorological Data. The following meteorological data will be required to support Shuttle operations:

a. Surface and upper air wind profiles

b. Ceiling and cloud cover

c. Visibility

d. Vertical temperature profiles

e. Humidity

f. Pressure

g. Density

h. Precipitation

i. Lightning potential

j. Turbulence

k. Storm location, intensity, movement

l. Sea state

m. Particles (hail, blowing dust/sand)

3.2.1.4.1.1.1 Conventional Civil and Military Meteorological Data. These data will be derived from normally scheduled conventional observations, analyses, and predictions such as:

a. Surface (aviation and synoptic) from U.S., foreign countries, and ships

b. Upper air (Rawinsonde, Radiosonde, and Rocketsonde Pibals) from US., foreign countries, and ships.

c. Weather radar

d. Aircraft pilot reports

e. Meteorological satellites

3.2.1.4.1.2 Space Environment Data. The following space environment data will be required to support Shuttle operations. These data will be derived from established solar observatories, operating satellites, and various other environmental and solar observing facilities.

3.2.1.4.1.2.1 Conventional Space Environment Data.

a. Solar Observation

Solar flare reports (e.g., size, location, time, region behavior, etc.)

Solar flare data (RF and X-ray background peak fluxes, times, etc.)

b. Geophysical and Interplanetary

Energetic particle reports

Artificial vent reports

3.2.1.4.2 DOD Security. The Shuttle System shall have the capability to process and secure classified STS mission data during any phase of operation, including mission planning, launch, flight, landing, post-landing, and turnaround. This includes STS mission data loaded into or residing in the Orbiter, simulators, and related ground equipment and facilities. The Orbiter onboard computers shall be capable of being declassified by using approved memory overwrite or erase procedures. Communications security measures shall conform to NASA/USAF Interagency Agreement for STS COMSEC, September 18, 1979.

3.2.1.4.3 Landing Site Support. For the early flights, the Orbiter vehicle shall have the capability and ground support for safe landings from orbit in daylight or darkness at the launch site (Kennedy Space Center, Florida) and the secondary landing site (Edwards AFB, California). When operational, the Orbiter vehicle shall have the capability and ground support for safe landings from orbit in daylight and darkness at one of the two launch sites (Kennedy Space Center, Florida, and Vandenberg AFB, California), or the secondary landing site (Edwards AFB, California). In addition, a number of non-Shuttle implemented contingency landing sites will be available throughout the Shuttle Program as needed to support quick returns from orbit. Payloads will be removed from the Orbiter prior to ferry operations. Payload handling, maintenance, and transportation after payload removal will be the responsibility of the payload agent. On a selected basis, subject to Level II approval, payloads may be ferried to the launch site in the Orbiter payload bay.

3.2.1.5 Mission Abort Operations Functions. *

3.2.1.5.1 Safe Mission Termination. The Shuttle System shall provide, by intact abort, the safe return of personnel, payload, and Orbiter. Intact abort consists of safe separation of the Orbiter from other vehicle elements and the safe landing of personnel, payload, and Orbiter on a runway.

3.2.1.5.1.1 Intact Abort. In addition to the requirements established in other sections of this document, the following requirements shall apply for intact abort.

a. The Shuttle System shall provide the capability for intact abort through all mission phases with a payload range from 0 to 65,000 lbs. for the failures listed in 3.2.1.5.1.3.

b. The Shuttle System shall provide the same fault tolerance during an intact abort as for normal flight operations except for the system (SSME, LRB or OMS) that caused the intact abort.

c. Higher TPS bondline temperatures following landing which may decrease the useful life of the vehicle shall be acceptable.

d. Orbiter down weights acceptable for mission planning shall be 211,000 pounds for EOM and 240,000 pounds for mission aborts (RTLS, TAL, AOA). Special assessments are required if these are exceeded and will be handled with waivers on a mission by mission basis. The maximum payload weight shall be based on the landing weights and the vehicle weight elements associated with the inert Orbiter, the Space Shuttle main engines, personnel, and onboard fluids, which constitute the total useful load.

e. Secondary and Contingency Landing Sites may be considered for Orbiter and personnel recovery. Secondary and primary contingency landing sites will include, as a minimum, ground support equipment to ensure crew, vehicle, and payload safety.

f. The payload shall not jeopardize the capability of the Orbiter to perform intact abort.

g. During an abort, provisions must be made to get the combined vehicle (Orbiter plus payload) center-of-gravity within the entry and landing limits stated in Paragraph 3.3.1.2.1.2.2 prior to the start of atmospheric flight. This requirement applies to aborts during ascent and from on-orbit.

h. The Orbiter vehicle shall have the capability of mission termination after orbit insertion and return to the launch or secondary/contingency landing site.

i. The Backup Flight System (BFS) shall support all intact abort modes (RTLS, TAL, AOA and ATO) whether the abort mode is selected prior to or subsequent to BFS engagement.

j. The Shuttle shall have the capability to withstand plume heating effects incurred while flying backwards during RTLS abort at free-stream pitot pressures that do not exceed 4 psf. RTLS trajectories shall be designed to keep pitot pressures within this limit.

3.2.1.5.1.2 Intact Abort Modes. The following intact abort modes shall be utilized in the event one of the failures listed in 3.2.1.5.1.3 occurs and may be used for other reasons than the intact abort failures listed in 3.2.1.5.1.3.

a. The Shuttle Flight Vehicle shall have the capability to continue ascent from LRB ignition through LRB separation. *

b. The Shuttle Flight Vehicle shall have continuous intact abort capability during ascent provided by Return to Launch Site (RTLS), Trans-oceanic Abort Landing (TAL), or an Abort-Once-Around (AOA) capabilities.

c. The TAL abort mode shall provide intact abort coverage between RTLS and AOA.

   1. An alternate TAL site shall be available for reselection any time prior to Abort Switch TAL selection and PBI commit to preclude a launch scrub in the event of unfavorable weather at the primary TAL site.

   2. An alternate TAL site shall be available for reselection in the event of a subsequent SSME or LRB engine failure while a TAL is in progress. *

   3. An alternate RTLS site shall be available for reselection in the event the primary RTLS site is experiencing unacceptable landing weather conditions.

d. The Shuttle vehicle shall have the capability of continuing the appropriately initiated 3 SSME abort mode for a flight subsystem in a fail-safe configuration (not including TPS, primary structure, pressure vessels, OMS, or RCS) should a single SSME or LRB engine subsequently have a partial or complete loss of thrust. *

3.2.1.5.1.3 Intact Abort Failures. Intact abort shall be provided for the following subsystems or systems failures. These failures shall be considered singly without combinations.

a. Complete or partial loss of thrust from one Orbiter main engine

b. Complete or partial loss of thrust from one LRB engine on each LRB. *

3.2.1.5.2 Contingency Aborts. Aborts caused by failures not included in the intact abort category shall be classified as a contingency abort. Intact abort capability is not required throughout the mission phases for this class of abort.

3.2.1.5.2.1 Contingency Abort Criteria. The following criteria shall apply for contingency abort:

a. Contingency aborts will not be used to determine hardware design criteria

b. The Orbiter's and SSME's usable lifetime may be degraded

c. Software and hardware impact may be allowed where feasible and cost effective, with specific approval

3.2.1.5.2.2 Contingency Abort Failures. The following conditions constitute contengency abort failures:

a. Loss of thrust from 2 or 3 SSMEs

b. SSME TVC failure(s)

c. LRB TVC failure(s) *

d. Premature Orbiter separation

e. Failure to separate LRB from Orbiter/ET

f. Loss of thrust from multiple LRB engines *

3.2.1.5.2.3 Contingency Abort Requirements. For possible use in contingency situations where mission completion or intact abort modes are not applicable, the Orbiter shall provide the capability to:

a. Manually initiate main engine or LRB engine cutoff at any time.

b. Manually initiate the ET mechanical separation sequence at any time.

c. Provide an abort downmoding capability (from ATO to AOA) to be effective post-MECO for sequential multiple-SSMEs-out.

d. Provide a manual single engine control capability (utilizing RCS augmentation and OMS propellant) for 2 SSMEs-out.

e. Provide a second trajectory shaping capability for 3-SSME-out entry (i.e., retain abort MECO data slots).

f. Provide a direct transfer capability to "alpha recovery and load relief" immediately following an exoatmospheric type * Orbiter/ET separation for a multiple-SSMEs or LRB engines out downrange ditching (i.e., direct transfer from MM104 to MM602).

g. Provide a manually initiated and terminated OMS/RCS propellant maximum rate depletion capability during powered flight and immediately following Orbiter/ET separation (allowing for control and utilizing existing propulsion systems) for multiple-SSMEs-out CG control.

h. Provide a MPS propellant exoatmospheric dump capability in RTLS and immediately following Orbiter/ET separation (utilizing the existing "on-orbit MPS LOX dump") for sequential multiple-SSMEs-out CG control.

i. Execute the Orbiter/ET contingency abort separation sequence in accordance with Paragraph 3.2.1.1.10.2.5 in both the primary and backup flight systems.

j. Provide an integrated/manual LRB engine/SSME control capability (utilizing LRB & SSME throttling) for 2 or more LRB engines out. *

3.2.1.5.2.4 Contingency Abort Modes. Within the criteria established in Paragraph 3.2.1.5.2.1, the following abort modes shall be utilized:

a. During First Stage Flight: Fast ET separation followed by ditching or continuation of ascent through LRB staging. *

b. During Second Stage Flight: Termination of main propulsion, ET separation, descent, and downrange ditching or landing.

3.2.1.5.3 Loss of Critical Function. A failure in a system or subsystem causing the loss of a "critical function" shall be eliminated from intact abort design and contingency abort categories by including appropriate safety margins or redundancy levels in the design.

3.2.1.5.3.1 Loss of Critical Function Failures.

a. ET rupture/explosion

b. LRB rupture/explosion

c. Major structural failure

d. Complete loss of guidance and/or control *

e. Loss of thrust from 1 LRB (all engines)

f. SSME or LRB TVC hardover

g. Failure to separation Orbiter from ET

h. Nozzle failure (SSME or LRB)

i. Premature LRB separation

j. Unacceptable loss of thrust from 3 or more LRB engines

3.2.1.5.4 Range Safety Flight Termination System. The Shuttle vehicle shall have a range safety flight termination system for all orbital flight tests and operational missions as required.

*

3.2.1.6 Ferry Mission Functions

3.2.1.6.1 Ferry. The Orbiter vehicle shall be capable of being ferried within the contiguous United States. On a selected basis, subject to Level II approval, payloads may be ferried to the launch site in the Orbiter payload bay.

3.2.1.6.2 Total weight and CG of the Orbiter (with payloads) in the ferry configuration shall be within the limits specified in Figure 3.2.1.6.2 (to be supplied).

3.2.1.6.3 Ferry flight shall be conducted in accordance with the following constraints:

a. Clear of visible moisture.

b. Light turbulence or less (as defined in the U.S. Flight Information Supplement).

c. Inflight temperature minimum +15 degrees F.

d. Electrical power to RCS heaters when ambient temperature is below 60 degrees F.

e. Ambient pressure minimum of 8 psia.

f. Drying of upfiring RCS thrusters (prior to next ferry flight) if rain accumulation exceeds 0.75 inches during ground period.

g. Structural restrictions as specified in NSTS 07700, Volume X, Section 4, Structural Restrictions for Orbiter Operational Flights; STS 8-0574. *

h. Capability for temperature conditioned cargo bay purge shall be provided at intermediate landing sites when specified in the Payload Integration Plan.

i. Capability shall be provided to operate coolant pumps inflight and at intermediate landing sites for payload water coolant loops mounted in the Orbiter cabin.

3.2.1.7 Transport System Element Functions. *

3.2.1.7.1 Delivery of System Elements to Using Site. The capability shall be provided to transport Shuttle vehicle elements and related support equipment from the site of manufacture to the launch and landing site. Such capability shall include initial Orbiter delivery by ferry flight.

3.2.1.8 Recycle Launch Facility Functions *

3.2.1.8.1 Launch Facility Turnaround Support. The launch complex, including support equipment and facilities, shall be refurbished and revalidated following each launch of a Shuttle vehicle. Turnaround operations shall support flight vehicle preparation and subsequent launch activities in a timeframe compatible with the traffic model.

3.2.1.9 Perform Rescue Operations Functions. (TBD) *

Table 3.2.1.1.6 Insertion Accuracy
Maximum Allowable One Sigma Dispersion of Actual State Vector at MECO

State	Position (NM)	Velocity (Ft/Sec)
Downrange	0.1	4.0
Crossrange	0.4	10.5
Vertical	0.15	4.5

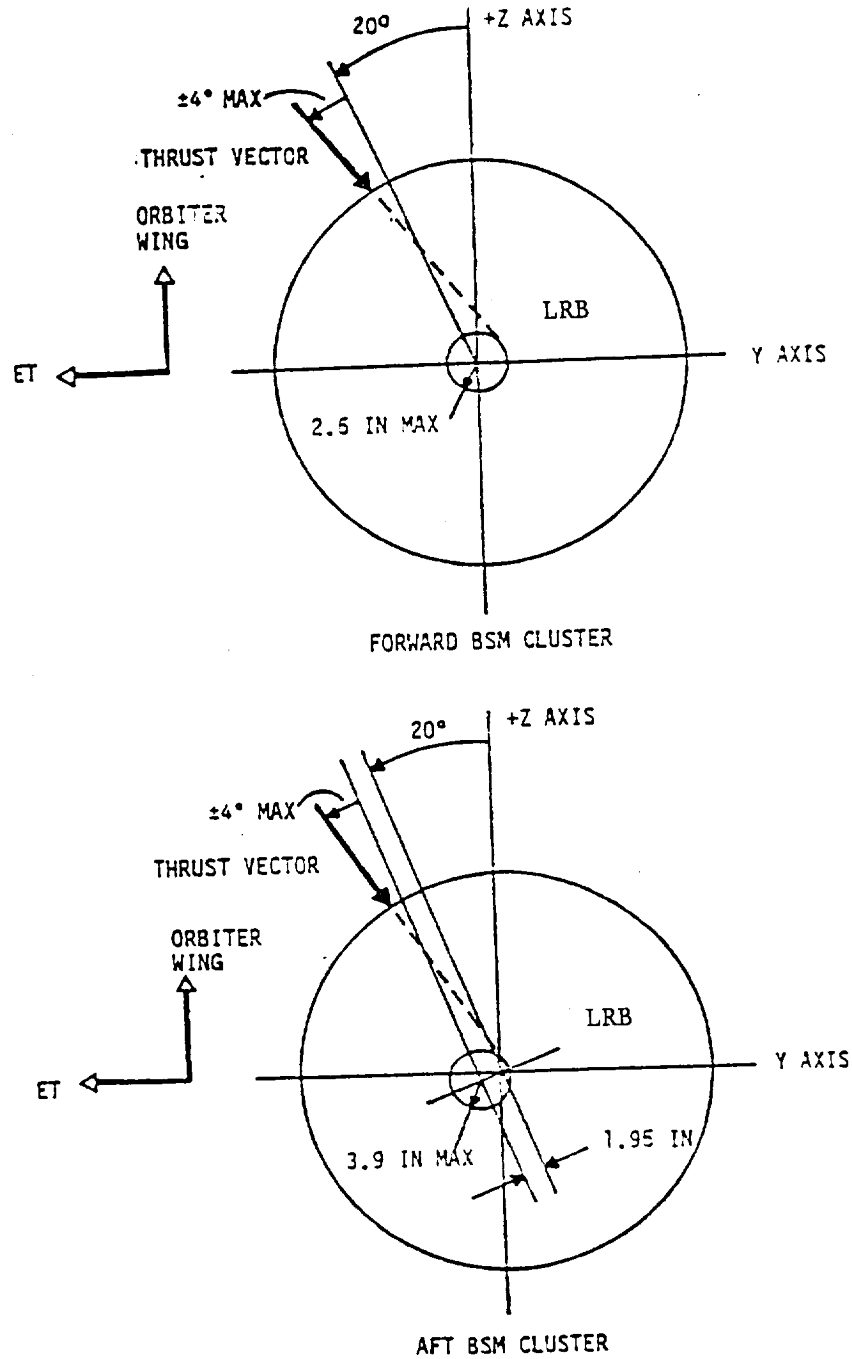


Figure 3.2.1.1.9.1.1.3 BSM Cluster Thrust Vector Orientation Tolerances

Parameter/Event	MECO	Structure Release	Separation Termination
Angle of Attack-Deg	-4 ± 2	-4 ± 2	10
Roll Angle-Deg	0 ± 2	0 ± 5	0 ± 30
Sideslip Angle-Deg	0 ± 2	0 ± 2	0 ± 3
Yaw Rate-$^{\circ}$/Sec)	$0 \pm .5$	$0 \pm .5$	0 ± 1
Roll Rate-$^{\circ}$/Sec)	$0 \pm .5$	0 ± 1.25	0 ± 2
Pitch Rate-$^{\circ}$/Sec)	$-25 \pm .5$	$-.25 \pm .5$	2.5 ± 2.5
Dynamic Pres.lb/ft^2		$0<\bar{q}\leq 10$	

Figure 3.2.1.1.10.2.4 Orbiter/ET Return to Launch Site Abort Design Staging Conditions

3.2.2 Design Characteristics

3.2.2.1 Flight Systems Design. The Shuttle System flight hardware shall consist of a reusable manned Orbiter vehicle including installed Space Shuttle Main Propulsion Engines (SSME), an expendable external tank, and Liquid Rocket Boosters (LRBs) which burn in parallel with the Orbiter SSMEs. The Orbiter vehicle shall be capable of crossrange maneuvering during entry and aerodynamic flight when returning from orbit. *

3.2.2.1.1 (Deleted).

3.2.2.1.2 Mated Ascent Guidance, Navigation, and Control. The Shuttle Flight Vehicle ascent guidance, navigation, and control function shall be accomplished in accordance with Paragraph 3.3.1.3.2.1. The Orbiter vehicle shall provide control to the Shuttle vehicle during mated ascent by throttling the MPS and/or LRB's to limit resulting rigid body, longitudinal acceleration as specified in 3.2.1.1.11. Aerodynamic, inertial, and thrust loads shall be limited by trajectory shaping and control, including throttling of the LRB and/or MPS, yaw steering, and elevon position changes for the following conditions: *

a. Design winds, shears, and gusts are as specified in NSTS 07700, Volume X, Appendix 10.10 applied with no SSME total thrust failures.

b. Design winds as specified in NSTS 07700, Volume X, Appendix 10.10 applied in conjunction with a total thrust loss from two LRB (one per LRB) engines and/or SSME. The dynamic effects due to gust penetrations and LRB engine and/or SSME thrust loss shall not be superimposed within five (5) seconds before or two (2) seconds after the failure occurs. *

The ascent flight control system shall provide the capability to parallel the SSMEs and LRBs in pitch and yaw axes during acceptable flight regions to enhance performance capability. The ascent flight control system shall also provide the capability to unparallel the SSMEs and LRBs to improve control authority and prevent engine collision as required. *

3.2.2.1.2.1 Shuttle Systems Avionics Terminal Events, Timing Contraints. The Shuttle System avionics terminal events times are contained in NSTS 07700 Volume X, Appendix 10.14 and TBD. The terminal event times are for: (1) SSME start command to LRB start command; (2) LRB start command to LRB ignition output command; (3) the LRB ignition command to LRB Holddown PIC fire output command; and (4) the LRB ignition command to T-0 umbilical retract PIC Fire output command. System timing includes both serial time dealys required for the initiation of a single event and the skew time between initiation of two events. *

3.2.2.1.2.2 Shuttle Systems Avionics Main Engine Shutdown events, Timing Constraints. The Shuttle Systems avionics timing constraints for Orbiter LO_2 prevalve close commands as referenced to either premature engine shutdown or MECO shutdown commands at the SSME controller interface are contained in Figure 3.2.2.1.2.2.

3.2.2.1.2.3 Lift-off Flight Control and Sequence. The ascent FCS shall initiate and execute the lift-off sequence and provide guidance and control to ensure that recontact between the mated vehicle and the launch facility is prohibited.

3.2.2.1.3 Aeroelasticity. Static and dynamic structural deformations and responses, including the effects of aeroelasticity under all limit conditions and environments, shall be accounted for in the structural design and shall not cause a system malfunction, preclude the stable control of the vehicle, or cause unintentional contact between adjacent bodies.

3.2.2.1.3.1 Static Aeroelasticity. The lifting surfaces shall be free from "divergence" and the aerodynamic control surfaces shall not exhibit "reversal" at dynamic pressures up to 1.32 times the maximum dynamic pressures, at the appropriate mach number, or boost, abort, entry, and aerodynamic flight envelopes.

3.2.2.1.3.2 Dynamic Aeroelasticity. The Shuttle vehicle shall be free from classical flutter, stall flutter, and control surface buzz at dynamic pressures up to 1.32 times the maximum dynamic pressure expected during flight. External panels shall be free of panel flutter at 1.5 times the local dynamic pressure at the appropriate temperature and mach number for all flight regimes including aborts.

3.2.2.1.4 POGO. The Space Shuttle Vehicle, in all mated and unmated configurations, shall be free of instabilities resulting from dynamic coupling of the structure, propulsion, and flight control subsystems during all phases of powered flight with all payload variations. Consideration will be given to stability margins, POGO suppression devices, OMS, LRB and main engine dynamic characteristics, the vehicle flight control subsystem, and appropriate parameter variations of these interacting subsystems. The total coupled system shall be stable for any allowable combination of system parameter variations. *

3.2.2.1.4.1 POGO Suppressor Requirements. A POGO suppressor shall be provided on each Space Shuttle main engine and LRB engine, if needed. The effective point of application of the suppressor shall be located on the SSME low pressure oxidizer turbopump discharge duct within 13 inches of the inlet flange of the high pressure oxidizer turbopump. The effective point of application for LRB engines is TBD. * *

3.2.2.1.4.1.1 Compliance (TBD) *

3.2.2.1.4.1.2 Inertance (TBD) *

3.2.2.1.4.1.3 Helium and Electrical Power Consumption. The suppressor design and operations shall minimize helium and electrical power consumption which will be supplied by the Orbiter.

3.2.2.1.5 Structure. The Shuttle vehicle structure, including pressure vessels and mechanical systems, shall have adequate strength and stiffness, at the design temperature, to withstand limit loads and pressures without loss of operational capability for the life of the vehicle and to withstand ultimate loads and pressures at design temperature without failure. The structure shall not be designed to withstand loads, pressures, or temperatures arising from malfunctions that prevent a successful abort. Major structural elements shall not be designed by nonflight conditions, i.e., conditions other than prelaunch (vehicle mating) through landing except for LRB water recovery if considered. *

3.2.2.1.5.1 Definitions. For the purpose of interpretation of this section, the following definitions will apply:

a. Limit Load. The maximum load expected on the structure during mission operation, including intact abort.

b. Ultimate Factor of Safety. The factor by which the limit load is multiplied to obtain the ultimate load.

c. Ultimate Load. The product of the limit load multiplied by the ultimate factor of safety.

d. Allowable Load. The maximum load which the structure can withstand without rupture or collapse.

e. Maximum Operating Pressure. The maximum pressure applied to the pressure vessel by the pressurizing system with the pressure regulators and relief valves at their upper limit, with the maximum regulator fluid flow rate, and including the effects of system environment such as vehicle acceleration and pressure transients.

f. Proof Pressure. The pressure to which production pressure vessels are subjected to fulfill the acceptance requirements of the customer, in order to give evidence of satisfactory workmanship and material quality. Proof pressure is the product of maximum operating pressure times the proof factor.

g. Margin of Safety. The ratio of allowable load to ultimate load minus one.

h. Safe-Life. A design criteria under which failure will not occur because of undetected flaws or damage during the specified service life of the vehicle; also, the period of time for which the integrity of the structure can be ensured in the expected operating environments.

3.2.2.1.5.2 Ultimate Factors of Safety. The ultimate factors of safety given in Table 3.2.2.1.5.2 shall be used for the Shuttle vehicle structure. The following specific conditions are allowed:

a. The ultimate factors of safety for LO_2 tank buckling shall not be less than 1.25 prior to initiation of prepressurization.

b. A safety factor of 1.491 for Power Reactant Storage Assembly is acceptable for PRSD tank unit Part No. MC282-0063-0100 S/N SX T0010.

3.2.2.1.5.3 Design Thickness. Stress calculations of structural members, critical for stability, shall use the mean drawing thickness or 1.05 times the minimum drawing thickness, whichever is less. Structural members, critical for strength, shall use the mean drawing thickness or 1.10 times the minimum drawing thickness, whichever is less.

3.2.2.1.6 Ultimate Combined Loads. The mechanical external, thermally induced and internal pressure loads should be combined in a rational manner. Any other loads induced in the structure, e.g., during manufacturing, shall be combined in a rational manner. In no case shall the ratio of the allowable load to the combined limit loads be less than the factor in Table 3.2.2.1.5.2.

K_1L external + K_1L thermal + K_2L pressure $\geq$ 1.40 Sigma L.

K_1 = 1.4 for boost conditions when the term is additive to the algebraic sum, Sigma L.

K_1 = 1.4 for entry, atmospheric cruise, and landing when the term is additive to the algebraic sum, Sigma L.

K_2 = 1.4 for the ET and LRB main propulsion tanks and SSME and LRB engines when the term is additive to the algebraic sum, Sigma L. *

K_2 = 1.5 for all other tankage when the term is additive to the algebraic sum, Sigma L.

K_1, K_2 = 1.0 when the term is subtractive to the algebraic sum, Sigma L.

L external = Mechanical externally applied loads, e.g., inertial loads, aerodynamic pressures.

L thermal = Thermally induced loads.

L pressure = Maximum relief valve setting where additive to algebraic sum, Sigma L.

= 0 to minimum regulated when subtractive to algebraic sum, Sigma L.

Ultimate load = $K_1 L_{external} + K_1 L_{thermal} + K_2 L_{pressure}$.

where: K_1 = The appropriate design factor of safety in Table 3.2.2.1.5.2 when the term is additive to the algebraic sum.

K_2 = The appropriate design factor of safety in Table 3.2.2.1.5.2 for all pressure vessels when the term is additive to the algebraic sum.

K_1, K_2 = 1.0 when the term is subtractive to the algebraic sum.

3.2.2.1.7 Allowable Mechanical Properties. Values for allowable mechanical properties of structural materials in their design environment, e.g., subjected to single or combined stresses, shall be taken from MIL-HDBK-5, MIL-HDBK-17, MIL-HDBK-23, or other sources approved by NASA. Where values for mechanical properties of new materials or joints, or existing materials or joints in new environments are not availabe, they shall be determined by analytical or test methods approved by NASA. Complete documentation of testing and analyses used to establish material properties and design allowables shall be maintained by the contractor, and the documentation shall be made available to the procuring agency on request. When using MIL-HDBK-5, material "A" allowable values shall be used in all applications where failure of a single load path would result in loss of vehicle structural integrity. Material "B" allowable values may be used in redundant structure in which the failure of a component would result in a safe redistribution of applied loads to other load-carrying members.

3.2.2.1.8 Fracture Control. In addition to the ultimate factors of safety presented in Paragraph 3.2.2.1.5.2, designs for primary structure, windows, glass components of other subsystems, and tanks shall consider the presence of sharp cracks, crack-like flaws, or other stress concentrations in determining the life of the structure for sustained loads and cyclic loads coupled with environmental effects. Parts determined to be fracture critical, including all pressure vessels*, shall be controlled in design, fabrication, test, and operation by a formal, NASA approved, fracture control plan as specified in SE-R-0006, "JSC Requirements for Materials and Processes".

*For the purpose of this paragraph, a pressure vessel is defined to be a component designed primarily for the storage of pressurized gases or liquids.

3.2.2.1.9 Fatigue. Safe life design shall be adopted for all major load-carrying structures. These structures shall be capable of surviving without failure a total number of mission cycles that is a minimum of four times greater than the total number of mission cycles expected in service (shown by analysis or by test through a rationally derived cyclic loading and temperature spectrum). This does not preclude fail-safe structural features.

3.2.2.1.10 Creep. The design shall preclude cumulative creep strain leading to rupture, detrimental deformation, or creep buckling of compression members during their service life. Analysis shall be supplemented by test to verify the creep characteristics for the critical combination of loads and temperatures.

3.2.2.1.11 Flight Vehicle Main Propulsion Propellants. The flight vehicle shall provide storage capacity for the main propulsion propellants in accordance with NSTS 07700, Volume X, Appendix 10.12. *

3.2.2.1.12 Tank/Liquid Flight Control Coupling. Tanks containing liquid and the flight control system shall be designed jointly to prevent or suppress coupling between the slosh of the liquid, the vehicle structure, and the flight control system.

3.2.2.1.13 Propellant Loading Accuracies. The root sum square overall system loading accuracy to the 100% mass load level of the Space Shuttle System shall be ± 0.43% for LO2 AND ± 0.35% FOR LH2. The allocation of uncertainties between the vehicle and ground system are given in Table 3.2.2.1.13.

THIS PAGE INTENTIONALLY LEFT BLANK

3.2.2.1.14 LO_2 Geyser Suppression. The ET and LRB LO_2 fill, drain, and engine feed design shall include provisions to suppress geysering to preclude damaging the ET, Orbiter, SSME and LRB.

3.2.2.1.15 ET and LRB Venting. *

3.2.2.1.15.1 O_2 Venting. The O_2 vent system shall not interface with the Orbiter but shall vent directly into the atmosphere. In addition to providing ET and LRB relief protection, the vent valves shall be capable of being actuated open, prior to launch, by ground command. The electrical command and pneumatic supply will be provided by GSE. Capability shall be provided to monitor the main propulsion LOX system pressures when vehicle or ground power are not applied to the flight instruments. *

3.2.2.1.15.2 Cryogenic Fuel Vent. (TBD) *

3.2.2.1.16 LO_2 Compatibility. Any material used internally in the liquid oxygen system of the Space Shuttle System propulsion subsystems shall be compatible as determined by NHB 8060.1. *

3.2.2.1.17 Design Environments.

3.2.2.1.17.1 Natural Environment. The Shuttle Flight Vehicle design shall satisfy the natural environment design requirements specified in NSTS 07700 Volume X, Appendix 10.10. *

3.2.2.1.17.2 Induced Environment. Each element of the Space Shuttle and its structural interfaces shall be capable of withstanding the incurred environment imposed during transportation, ground operations, and flight operations as defined in NSTS 07700, Volume X, Appendix 10.11. *

3.2.2.1.17.2.1 Ascent Heating Design Criteria. In general, all elements of the Space Shuttle System shall be designed to withstand limiting induced ascent aerodynamic and plume heating environments, encompassing all baseline reference missions. The Orbiter vehicle for which limit ascent aerodynamic heating environments coupled with reuse criteria would result in unnecessary weight and cost penalties, shall be designed to meet reuse requirements considering the frequency of occurence of the ascent heating environments resulting from statistical treatment of the baseline reference missions and shall be shown to have single mission survivability for limit ascent aerodynamic heating case encountered on any mission during the lifetime of the vehicle. The applicable environments are defined in NSTS 07700, Volume X, Appendix 10.11. *

3.2.2.1.17.3 TPS Absorption. All TPS material and installation design shall minimize absorption and entrapment of liquids or gases which would degrade * thermal or physical performance or create a fire hazard (wicking), and shall not require draining or drying. A dedicated purge system shall not be required from refurbishment through launch, except for inadvertent exposure to rain after flight and before weatherproofing.

3.2.2.1.18 Flow Induced Vibration. All flexible hoses and bellows shall be designed to exclude or minimize flow induced vibrations in accordance with MSFC-DWG-20M02540. Certification of hardware shall be in accordance with NSTS 08123.

3.2.2.1.19 Cross Contamination. Cross contamination of Space Shuttle System elements, such as LRB jettisoning engine plume impingement of the Orbiter, * shall be minimized.

3.2.2.1.20 Flight Element Mating Design Characteristics

3.2.2.1.20.1 ET/LRB Joints. The joint concept for ET/LRB utilization shall * be capable of:

a. Assembly without internal access to the ET.

b. Assembly with access sufficient to

   1. Easily verify alignment of mating interface;

   2. Easily join (with positive engagement) the mating joint.

c. Assembly without requirement for makeup of explosive devices during mating.

d. Assembly allowing use of a nominal "0" moment joint in the ET interstage.

e. Assembly allowing unrestrained rotation in the Orbiter/ET plane.

f. Assembly within the operational timeline.

g. Accommodating shrinkable induced loads caused by ET or LRB cryogen loading and LRB expansion or contraction. *

h. Restricting LRB pitch misalignments, both LRBs deflected symmetrically, to ± 0.25 degree maximum during launch and boost flight, for aerodynamic performance and flight control considerations. *

i. Restricting LRB yaw misalignments to ± 0.25 degree maximum during launch and boost flight for aerodynamic performance and flight control considerations. *

3.2.2.1.20.2 Orbiter/ET Joints. The joint concept for Orbiter/ET utilization shall be capable of:

a. Restricting Orbiter pitch misalignment to ±0.25 degree maximum during launch and mated flight operations for aerodynamic performance and flight control considerations.

b. Restricting Orbiter yaw misalignments to ±0.25 degree maximum during launch and mated flight operations for aerodynamic performance and flight control considerations.

3.2.2.1.21 Instrumentation Calibration Data. The flight system instrumentation shall condition signal output data to follow specified theoretical or model characteric curves. The output signal shall be within a specified error band of the characteristic curve. Deviation from this standard shall be allowed only where necessary to meet instrumentation accuracy requirements.

3.2.2.1.22 Flight Termination System Design. A ground-commanded flight termination system shall be provided on LRBs to destruct the LRBs, and on the ET to disperse ET propellants. LRB system components shall be reusable where * cost savings will result. The ET design shall provide flexibility by addition or removal of components where possible. The system shall not require any action by the crew to operate.

3.2.2.1.22.1 (Deleted).

3.2.2.1.22.2 Destruct Safing. The LRB destruct systems shall be safed electronically and mechanically prior to normal LRB separation by an automatic signal from the Orbiter so that destruct action cannot occur and the LRBs are * safe for recovery/retrieval operations. The mechanical safing shall provide a physical interruption of the ordnance train.

3.2.2.1.22.3 Command System. The flight termination system radio command system shall utilize a separate, secure flight code for each ARM and fire command so configured that continuous transmission of unauthorized correctly structured random formats for 30 minutes would allow not more than 1 chance in 10^6 of a valid command being accepted. The operational ground and flight codes shall be classified and preflight testing shall be accomplished without radiating the operational codes.

3.2.2.1.22.4 Range Safety Abort Light. Receipt of an "ARM" command by the range safety flight termination system shall illuminate an Orbiter display light to warn the crew.

3.2.2.1.22.5 Real-time Telemetry. The Orbiter shall provide real-time RF transmission of range safety system (RSS) telemetry parameters through ascent.

3.2.2.2 Ground System Design. The ground system shall be designed to withstand or be protected from the effects of the natural environments defined in NSTS * 07700, Volume x, Appendix 10.10 in addition to the requirements outlined in the following paragraphs.

3.2.2.2.1 Ground Facilities. New ground facilities shall be designed in accordance with NHB 7320.1, Facilities Engineering Handbook.

3.2.2.2.2 Ground Support Equipment. GSE required by the Shuttle Ground Operations Systems shall be designed in accordance with SW-E-0002, Ground Support Equipment, General Design requirements.

3.2.2.2.3 (Deleted).

3.2.2.2.4 (Deleted).

3.2.2.2.5 (Deleted).

3.2.2.2.6 GSE Control and Monitoring. When hazardous operations or safety dictates, servicing equipment and GSE used during test and launch operations shall interface with ground stations to provide control and status monitoring.

Table 3.2.2.1.5.2 Ultimate Factors of Safety

Components	Factors of Safety (Ultimate)
General structure & main propellant tanks	≥ 1.40 (A) (G) (H) (I) (J) (B)
Pressurized windows	
A. Annealed panes	
Initial F.S.	≥ 2.0
Final F.S.	≥ 1.0
B. Tempered panes	
Initial F.S.	≥ 2.0
Final F.S.	≥ 2.0
Pressurized manned compartments	1.5
Pressure alone	1.5
Main propellant tanks ET & LRB (pressure alone)	--- (C) (H)
Pressure vessels (other than main propellant tanks)	≥ 1.5 (A) (B)
Pressurized lines and fittings	
Less than 1.5-in. dia	4.0 (E) (F)
1.5-in. dia or greater	1.5

(A) See Paragraph 3.2.2.1.6

(B) See Paragraph 3.2.2.1.8

(C) Factor of safety specified in element CEI and as determined by Paragraph 3.2.2.1.8

(D) (Deleted)

(E) Design of hydraulic systems shall be in accordance with MIL-H-5440

(F) Lines and fittings of less than 1.5-in. diameter may be designed to a minimum factor of safety of 1.5 where advantageous to the Shuttle vehicle, providing the rigor of design analysis and verification testing performed is equivalent to that applied to other critical systems/components. Whenever the exception allowed by this Paragraph is utilized by an element, the affected system/components shall be identified along with a brief descriptiion of the analysis and testing applied to justify the adequacy and acceptability of the lower factor of safety. All exceptions must be approved by the Program Manager.

(G) The landing gear system design shall comply with the following structural loads design criteria:

Table 3.2.2.1.5.2 Ultimate Factors of Safety - Concluded

Loading Condition	Loads Definition	FS	Material Allowable
Landing Touchdown Loads	Design	1.0	Yield
Rollout and Ground Handling	Limit	$\geq$1.4	Ultimate

* From MIL-A-8862, Paragraph 3.1.3

(H) LRB general structure - Before LRB separation, ultimate factor of safety = 1.40. Exceptions to this requirement are (TBD) *
and will be handled on a case-by-case basis.

(I) For the ET, the factor of safety for highly predictable quasi-static loads shall be equal to or greater than 1.25. Examples of such loads are steady thrust, inertial loads from steady acceleration and weight.

(J) The ultimate FS of the LRB/ET forward separation bolt fracture groove shall be $\geq$ 1.34. The 1.34 factor of safety is based on a maximum tensile load of -189,100 lbs. *

Table 3.2.2.1.13 Propellant Loading Accuracies

*

TBD

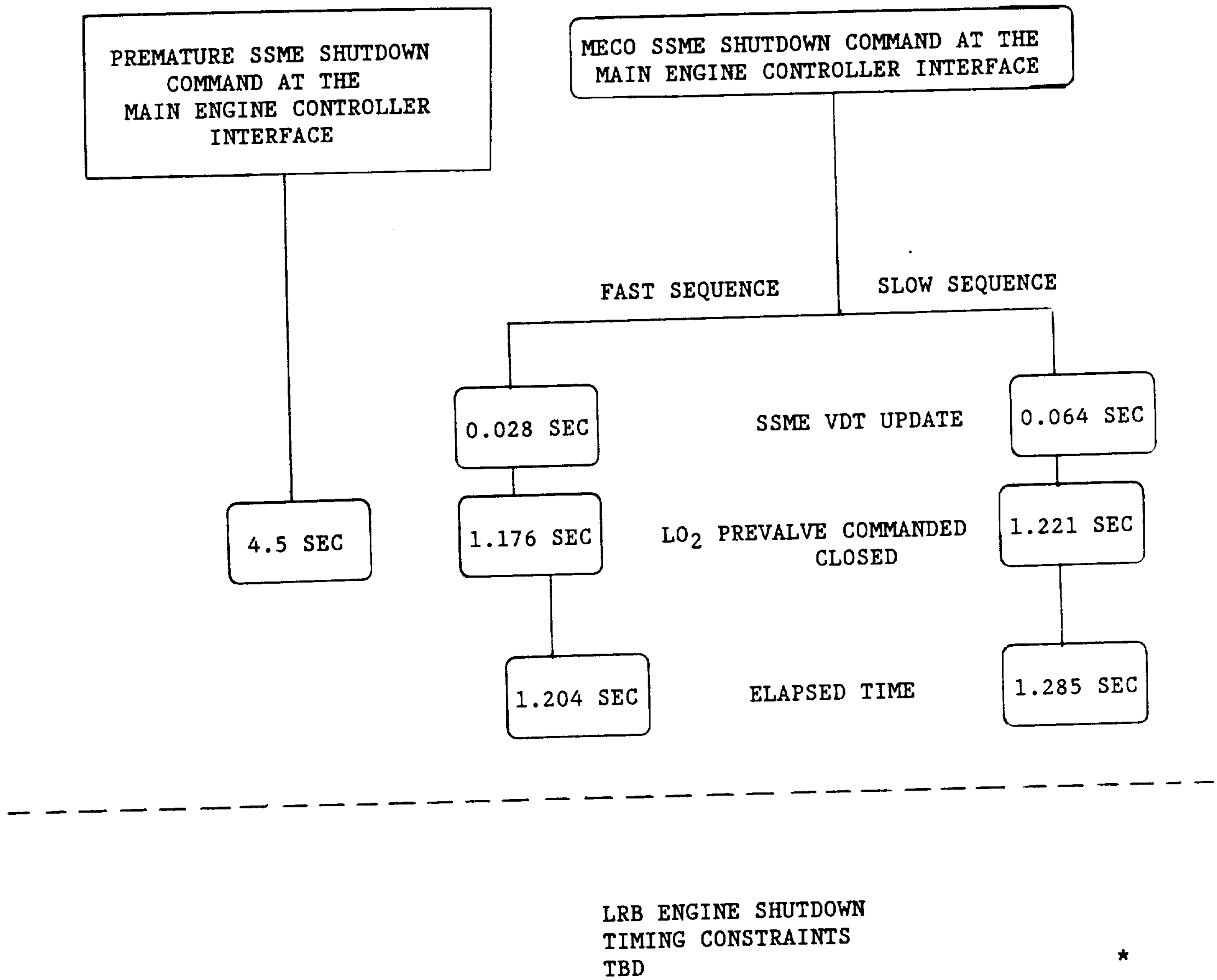


Figure 3.2.2.1.2.2 Shuttle Systems Avionics Main Engine and LRB Engine Shutdown Events, Timing Constraints

3.2.3 Logistics. Shuttle System logistics requirements are specified in NSTS 07700, Volume XII.

3.2.4 Personnel and Training. Shuttle System personnel and training requirements are specified in NSTS 07700, Volume XII.

3.2.5 Shuttle System Interface Characteristics

3.2.5.1 Shuttle System Interface with Communcations and Tracking Functions. The Shuttle System shall interface with the Communications and Tracking Functions as defined in the following ICDs:

a. Space Shuttle System/KSC RF Communcation and Tracking ICD 2-0A004

b. Shuttle Communications and Tracking/USAF ICD 2-0D003

c. Shuttle Communications and Tracking/STDN ICD 2-0D004.

3.2.5.1.1 Range Safety Real-time Data. Specific telemetered Shuttle position data and systems measurements will be provided to the AFETR range safety officer's facility in realtime from prelaunch through ascent.

3.2.6 Shuttle System Measurement Requirements

3.2.6.1 Master Measurement List for Space Transportation System. (TBD) *

3.2.6.1.1 Shuttle Orbital Flight Test Calibration Data Plan. The Orbital Flight Test Calibration Data Plan is specified in JSC 13047.

3.2.6.2 Main Propulsion Test Article Master Measurements List. The Main Propulsion Test Article Master Measurements List is specified in NSTS 08222.

3.2.7 Test Requirements and Specifications for the Shuttle Main Propulsion Test Program.

3.2.7.1 Test Requirements and Specifications Document. The test requirements and specifications for MPT are specified in NSTS 08200.

3.2.8 Test Requirements and Specifications for the Shuttle Mated Vertical Ground Vibration Test Program.

3.2.8.1 Test Requirements and Specifications Document. The test requirements and specifications for the MVGVT are specified in NSTS 08201.

3.2.9 Test Measurements List for the Shuttle Mated Vertical Ground Vibration Test Program.

3.2.9.1 Test Measurements List. The test measurements for the MVGVT are specified in JSC 08223.

3.2.10 Operations and Maintenance Requirements and Specifications.

3.2.10.1 Operations and Maintenance Requirements and Specifications. The Operations and Maintenance Requirements and Specifications for the Orbiter are specified in JSC 08171, File I.

3.2.11 Operations and Maintenance Requirements and Specifications for the Orbital Flight Test for the Space Shuttle Program.

3.2.11.1 Operations and Maintenance Requirements and Specifications Document. The Operations and Maintenance Requirements for the Orbiter are specified in JSC 08171, File III, Volumes 1, 2, 3, 4, 5, 6, and 7.

3.2.12 Operations and Maintenance Requirements and Specifications for the External Tank for the Space Shuttle Program.

3.2.12.1 Operations and Maintenance Requirements and Specifications Document. The operations and maintenance requirements for the External Tank are specified in JSC 08171, File IV.

3.2.13 Operations and Maintenance Requirements and Specifications for the Integrated OMRSD for the Space Shuttle Program.

3.2.13.1 Operations and Maintenance Requirements and Specifications Document. The Integrated OMRSD is specified in JSC 08171, File II, Volume 1.

3.2.14 Operations and Maintenance Requirements and Specifications for the Liquid Rocket Booster for the Space Shuttle Program. *

3.2.14.1 TBD *

3.2.15 Operations and Maintenance requirements and Specifications for the Orbital Flight Test for the Space Shuttle Program.

3.2.15.1 Operations and Maintenance Requirements and Specifications Document. The operations and maintenance requirements for the Primary Flight Control System End-to-End Accuracy Test is specified in JSC 08171, File III, Volume 9.

3.2.16 Operations and Maintenance Requirements and Specifications for the Orbiter Integrated Tests for the Space Shuttle Program.

3.2.16.1 Operations and Maintenance Requirements and Specifications Document. The operations and maintenance requirements for the Orbiter Integrated Test is specified in JSC 08171, File III, Volume 8.

3.2.17 Operations and Maintenance Requirements and Specifications for the Orbiter Control Loop Dynamic Stability Test for the Space Shuttle Program.

3.2.17.1 Operations and Maintenance Requirements and Specifications Document. The operations and maintenance requirements for the Orbiter Control Loop Dynamic Stability Test is specified in JSC 08171, File III, Volume X.

3.2.18 Operations and Maintenance Requirements and Specifications for Mission Equipment Kits for the Space Shuttle Program.

3.2.18.1 Operations and Maintenance Requirements and Specification Document. The operations and maintenance requirements for mission kits are specified in the appropriate flight vehicle element file of JSC 08171.

3.2.19 Test Requirements and Implementation Plan for KSC MLP HDP Stiffness.

3.2.19.1 Test Requirements and Implementation Plan for KSC MLP HDP Stiffness Verification. The test requirements and implementation plan for the KSC MLP HDP stiffness verification are documented in NSTS 08206.

3.2.20 Operations and Maintenance Requirements and Specifications for Ground Support Equipment for the Space Shuttle Program.

3.2.20.1 Operations and Maintenance Requirements Specifications Document. The Operations and Maintenance Requirements for Ground Support Equipment are specified in JSC 08171, File VI for KSC and File XII for VLS.

3.3 SHUTTLE VEHICLE END ITEM PERFORMANCE AND DESIGN CHARACTERISTICS.

3.3.1 Orbiter Vehicle Characteristics.

3.3.1.1 Orbiter Performance Characteristics.

3.3.1.1.1 Pointing Accuracy. For payload pointing purposes, the Orbiter vehicle shall be capable of attaining and maintaining any desired inertial local verticle, earth surface pointing, or orbital object pointing attitude within the thermal constraints defined in Section 3.2.1.1.12. The GN&C shall have the capability to point any vector defined in either the IMU navigation base axis system or the corresponding axis system of an equally accurate payload supplied and payload mounted sensor (see Paragraph 3.3.1.3.3.5.2) to within +0.5 degrees of the desired attitude (other than for orbital object pointing). For payload pointing utilizing the Vernier RCS, the Orbiter Flight Control System (FCS) shall provide a stability (deadband) of $\pm$0.1 deg/axis and a stability rate of (maximum limit cycle rate) of $\pm$0.01 deg/sec/axis when no vernier RCS thrusters are failed. When using the large RCS thrusters, the Orbiter FCS shall be capable of providing a stability of $\pm$0.1 deg/axis and a stability rate of $\pm$0.2 deg/sec/axis. Propellant needed for this requirement shall be chargeable to payload weight (see Paragraph 3.2.1.1.2.1.1). For payload pointing and/or stability requirements beyond the capability of the Orbiter, the Orbiter shall be capable of interfacing with a payload-supplied and payload-mounted stabilization and control system.

3.3.1.1.2 Rendezvous with Cooperative Target. The Orbiter vehicle shall have an onboard capability to rendezvous with nominally inplane cooperative targets and shall be the active vehicle during rendezvous, docking, and undocking. Maximum RF tracking range shall be at least 300 nm.

3.3.1.1.3 Rendezvous with Passive Target. The Orbiter vehicle, by using ephemeris data from ground facilities and onboard sensors and computation, shall be capable of rendezvous with a passive, stabilized orbiting element. Onboard RF tracking sensor information shall be provided for ranges equal to or less than 19 km for a target whose effective RF cross-sectional area is 1 square meter. The passive target sensor weight shall be a part of the Orbiter and shall assess no weight or volume penalty to the payload.

3.3.1.1.4 Crew Controlled Docking. The Orbiter vehicle shall be capable of crew controlled docking to other Orbiter vehicle(s) or other compatible orbiting elements during daylight or darkness.

3.3.1.1.5 Ranging Requirements. The Orbiter vehicle shall have the crossrange capability to return to any pre-mission selected nominal, alternate, or intact abort landing site supported by NSTS 07700, Vol X, Appendix 10.17 (Requirements for Runways and Navigation Aids) for orbital inclinations from 28.5 to 104 * degrees, inclusive. Provision shall be made for downrange maneuver capability accounting for the effects of deorbit, entry guidance and entry dispersions, including navigation, aerodynamic, atmospheric, and weight uncertainties. The capability shall be consistent with that which is achievable from the program baselined entry angle of attack profile of 40.0 degrees from any orbit.

3.3.1.1.6 Payload Integration. The Shuttle System shall provide for payload removal or installation including replacement of removed payloads by dissimilar payloads with the Orbiter vehicle in the vertical position on the launch pad. Vertical installation/removal reconfiguration capability shall be provided for the following Orbiter payload bay flight kits:

a. Payload structural attachments.

b. Standard Mixed Cargo Harness (SMCH) and Spacelab harness.

c. Mid and aft fuselage ballast.

d. OMS delta V propulsion module (requires 3-point longeron attachment).

e. T-4 hours payload umbilical (requires payload removal above umbilical)

f. Rescue.

At all other locations, payloads shall be installed and removed with the Orbiter vehicle in the horizontal position. Both of the installation operations (horizontal or vertical) shall be considered baseline and each consistent with the current timeline allocations. When required for a specific mission and in conjunction with a compatible payload, the installation and/or removal of an OMS delta V propulsion module shall be possible without impact to or by an installed payload. Payloads will have a standard ground handling interface.

3.3.1.1.7 Payload Bay/Payload Access.

a. Physical Access in Orbiter Processing Facility Station:

   The Orbiter vehicle and ground system facility shall provide access to the payload bay and the payload through the Orbiter crew compartment and by opening the payload bay doors.

b. Physical Access in VAB Integration Cell Station:

   The Orbiter vehicle and ground system facility shall provide access to the payload bay and the payload through the Orbiter crew compartment. Any special provisions for personnel access in the payload bay shall be provided by payloads.

c. Physical Access at Launch Pad Station:

1. Horizontally Installed Payloads - The Orbiter vehicle and payload shall provide access to the payload through the Orbiter crew compartment until start of crew compartment closeout at approximately T-12 hours. Any special provisions for personnel access in the payload bay shall be provided by payloads.

2. Vertically Installed Payloads - The Orbiter vehicle and ground facility shall provide access to the payload through the payload bay doors until start of payload bay door closing prior to hypergol servicing at approximately T-13 hours.

3. Late Access to Payload Bay - Late access for unique servicing and adjustment of payload elements after hypergol servicing operations shall be possible through the payload bay doors. This will increase the pad timeline and physical access to the payload bay would terminate 8-1/2 hours prior to lift-off.

d. Physical Access at the Landing Site Stations:

The Orbiter vehicle and ground system facility shall provide access to the payload bay through the crew compartment from landing to landing plus one hour. Any special provisions for personnel access in the payload bay after landing shall be provided by payloads. Access through the payload bay doors will be available in the Orbiter Processing Facility Station, compatible with Orbiter safing operations (approximately landing plus 16 hours).

Note: Payload and payload bay access requirements not satisfied by the above criteria will require unique Shuttle operational scheduling and an increase in the 160-hour baseline allocations.

3.3.1.1.8 Orbiter Landing. For Orbiter return, the Orbiter vehicle shall be capable of operating into airfields that have runways equivalent to 12,500 feet long and 150 feet wide at sea level on a hot day (103°F). The Orbiter * vehicle shall be designed to land on such runways, allowing for hot temperatures, wet grooved surfaces, and the wind conditions specified in NSTS 07700, Volume X, Appendix 10.10. NSTS 08192 defines the Math Model of Friction Characteristics for Orbiter Main and Nose Gear Tires. The Orbiter shall also have the capability to land under manual control.

3.3.1.1.9 Payload Weight at Landing. The Orbiter vehicle shall be designed to land 32,000 lbs. of payload within the design load factors with the environmental conditions specified in NSTS 07700, Volume X, Appendix 10.10. Landings with heavier payloads (up to 65,000 lbs.) shall be constrained by an Orbiter maximum landing weight of 240,000 lbs. as specified in Paragraph 3.2.1.5.1.1d. Propellant dump provisions or other provisions may be necessary to accommodate these constraints for the various abort cases, RTLS, TAL, AOA, and AFO.

3.3.1.1.10 Extended Missions. The Orbiter vehicle design shall not preclude the capability to extend the orbital stay time up to a total of 30 days. This requirement shall not affect the cabin size or expendables, as defined in 3.2.1.1.2.1.10.

3.3.1.1.11 Passive Control Mode. For those payload experiment operations requiring essentially zero g translational accelerations with no attendant pointing requirements, the Orbiter shall be capable of operating in a passive (free drift with jets inhibited) control mode within the Orbiter thermal constraints defined in Section 3.2.1.1.12.

3.3.1.1.12 Payload Deployment Operations. During payload deployment operations, the payload will be capable of sustaining loads imposed by the Shuttle as a result of RCS attitude control. The definition of the flight control system for RCS attitude control will be contained within the appropriate PIP or ICD. Payload deployment operations include erection and/or extension. This requirement does not apply to payloads requiring the RMS for payload deployment/handling operations. Refer to NSTS 07700, Volume XIV, Paragraph 8.1.1 for RMS requirements.

3.3.1.2 Orbiter Design Characteristics.

3.3.1.2.1 Structure and Mechanical Subsystems.

3.3.1.2.1.1 Cabin Size. The cabin shall be designed to accommodate a total crew of seven: three crewmen to operate the Orbiter and up to four payload specialists. The design shall not preclude installation of crew support equipment for a total of 10 crew members as required to implement an Orbiter-to-Orbiter rescue. Configuration of the panels and structure above and below the interdeck access hatch shall permit passage of a crewman in a pressurized EMU. Shuttle System weight and performance control shall be per Volume X, Section 3.1.3.

3.3.1.2.1.2 Payload Accommodations.

3.3.1.2.1.2.1 Payload Envelope. A clear payload envelope 15 feet in diameter and 60 feet in length shall be provided in the Orbiter payload bay. Payload thermal and dynamic deflections and all payload protrusions (except the payload attachment fittings) shall be contained within the payload envelope. Payload side attachment fittings shall extend beyond the payload envelop to mate with the Orbiter side attachment fittings which are outside the payload envelope. Umbilicals required to interface the payload to the Orbiter or to GSE while the payload is in the payload bay may also penetrate the payload envelope.

3.3.1.2.1.2.1.1 Payload Bay Clearance. The clearance between the payload envelope and the Orbiter vehicle structure and subsystems shall be provided by the Orbiter. This clearance will prevent interference between the payload and Orbiter due to Orbiter deflection caused by the induced environment and during payload deployment. Payloads are constrained to the payload envelope, when subjected to the induced environment during the complete mission, beginning with payload installation and ending with payload deployment or removal.

3.3.1.2.1.2.1.2 Payload Viewing. The Orbiter shall have the capability of exposing the entire length and full width of the payload bay. With the payload door(s) and radiator(s) open, an unobstructed 180^{o} lateral field of view shall be available above the payload bay door frame and associated mechanism.

3.3.1.2.1.2.2 Payload Center-of-Gravity. The Orbiter vehicle shall provide an allowable center-of-gravity envelope as follows:

X CG (As shown in Figure 3.3.1.2.1.2.2 and tabulated in Table 3.3.1.2.1.2.2)

Y CG (As shown in Figure 3.3.1.2.1.2.2a and tabulated in Table 3.3.1.2.1.2.2a)

Z CG (As shown in Figure 3.3.1.2.1.2.2b and tabulated in Table 3.3.1.2.1.2.2b)

The payload center-of-gravity for payload weights up to 65,000 pounds must be within the specified envelopes at the time of MECO for RTLS abort, and at the time of entry (400,000 ft. altitude) for all other intact abort flightmodes. For normal mission flights, the payload center-of-gravity for payload weights up to 32,000 pounds must be within that specified portion of the envelopes at the time of entry (400,000 ft. altitude). The payload weight associated with the center-of-gravity envelopes includes all payload weight chargeable items required for the mission and location of such items. Even if they are located outside of the payload bay clearance envelope must be included in the payload CG determination.

3.3.1.2.1.2.3 Payload/Vehicle Dynamic Interfaces. Payload/vehicle dynamic interactions shall be minimized through proper design procedures. The Orbiter/payload combination shall be based on the payload frequency constraints contained in Volume XIV, Attachment 1 (ICD 2-19001).

3.3.1.2.1.3.4 Ku-Band Radiation Environment. The maximum Orbiter generated RF field intensity on payloads during orbital operations shall not exceed 68 volts/meter. The Orbiter shall provide positive limits on the Ku-Band antenna to preclude the irradiation of payloads in the cargo bay in excess of this field intensity. An override capability will be provided for missions with payloads which have a high maximum power density limitation (greater than 304 volts per meter). During deployment and retrieval, operational procedures will be used to ensure the RF field intensity limit is not exceeded. Payloads which cannot tolerate 68 volts/meter will require special provisions.

3.3.1.2.1.3 Docking Module. An androgynous docking module shall allow positive interception, engagement, and release of the Orbiter vehicle with other orbital elements or another Orbiter vehicle. It shall not be necessary to remove any part of the docking module to allow personnel or cargo transfer. A clear transfer passageway of not less than 0.92 meters diameter shall be provided. The docking module shall be removable when not required for the mission. The docking module shall be chargeable to the payload.

3.3.1.2.1.4 Payload Deployment and Retrieval mechanism. The Orbiter vehicle shall provide a payload deployment and retrieval mechanism which shall be chargeable to the Orbiter weight and shall be stowed outside the 60-foot length by 15-foot diameter payload enveloped. The deployment/retrieval mechanism shall have the capability to deploy and retrieve payloads with dimensions of 15-ft diameter and 60-foot length within two orbits after initiation of the sequence and in accordance with the requirements of Paragraphs 3.2.1.1.3 and 3.2.1.1.3.1. The deployment/retrieval mechanism shall have the capability to perform the deployment of a payload within 25 minutes from the release of the payload from holddown to release of the payload from the manipulator in space. The deployment/retrieval mechanism shall be utilized for zero g handling of payloads. Space orbiting elements may be berthed to the Orbiter using the deployment and retrieval subsystem. If tilt tables or swingout systems are requirement for payload handling, these devices shall be part of the payload, chargeable to payload weight and volume.

The Orbiter remote manipulator system shall:

a. Provide the manipulator arm on the Orbiter as standard equipment;

b. Provide scar weight and control mountings for a second manipulator arm as a payload option.

The capability shall exist to remove the one manipulator arm provided by the Orbiter, when not required, to provide additional payload weight capabilities.

The second manipulator system shall be installed as a payload weight chargeable kit. Capability shall be provided to operate two manipulators in serial-only (non-simultaneous) operations. Capability will be provided, however, to hold or lock the payload with one manipulator while operating the second manipulator arm. The Orbiter shall provide the capability to jettison and verify jettison of manipulator arm assemblies. The capability shall be provided to individually jettison each manipulator arm.

3.3.1.2.1.4.1 Multiple Payload Deployment and Retrieval. Within the reach limits of the deployment/retrieval mechanism, the Orbiter vehicle shall have the capability to deploy and retrieve single or multiple (5) payload elements on-orbit during a single mission, including placement or docking of payloads to a stabilized body. In applying this requirement, payload attachment schemes shall be compatible with Orbiter capability for operating a maximum of 15 active retention mechanism latches.

3.3.1.2.1.4.2 Payload Retention. An active retention and release mechanism for the payload shall be provided.

3.3.1.2.1.4.3 Payload Swingtable Attachment. Attachment of the payload swingtable shall be provided through use of existing payload attachments and/or hardpoints in the Payload Bay.

3.3.1.2.1.4.4 Payload contingency Retrieval. The payload deployment and retrieval mechanism shall have the capability to retrieve deployed/detached payloads, in a non time-constrained manner, weighing up to 65,000 pounds and suitably configured for Orbiter installation.

3.3.1.2.1.5 Orbiter Control Weight and CG Limits. The Orbiter element inert control weight is specified in NSTS 07700 Volume X, Appendix 10.12. The longitudinal CG of the operational Orbiter vehicle (with main engines), crew and provisions, and payload for entry through landing, be within the limits of 65% to 67.5% of the Orbiter body length. The Orbiter lateral CG variation shall be ±1.5-inches maximum, and the Orbiter vertical CG shall be between waterline stations 360.0 and 384.5 (Orbiter coordinates) upon entry.

The longitudinal CG of the operational Orbiter vehicle with main engines included, crew and provisions, and payload shall be within the limits of 65% to 67.5% of the body length.

3.3.1.2.1.6 TPS Design. The TPS shall be designed to accomplish the reference missions in 3.2.1.1.3. On-orbit thermal conditioning for up to 12 hrs prior to entry shall be accommodated for missions where the TPS temperatures exceed the design values associated with one revolution missions. For emergency entry, where preentry thermal conditioning cannot be performed, structure over temperature from design values is allowed with the resulting degradation in vehicle service life.

3.3.1.2.1.7 Orbital External Configuration. The Orbiter shall conform to the moldline envelope specified in TBD , "Shuttle Moldline and Protuberances". *

3.3.1.2.1.8 Airlock. An airlock shall be provided to accommodate two-man EVA operations without the necessity for crew cabin decompression, or decompression of an attached manned payload. The airlock shall accommodate any of the following installation configurations:

a. Inside the crew module.

b. Inside the crew module with tunnel adapter in series.

c. In the payload bay mounted on the aft side of the crew cabin bulkhead.

d. On top of the tunnel adapter in the payload bay.

3.3.1.2.1.9 Tunnel Adapter. A removable tunnel adapter with capability for attachment of an outside airlock shall be provided to accommodate continuous crew cabin to manned payload access during EVA. The tunnel adapter design shall allow an EVA crewman to access a depressurized Spacelab without the necessity of crew cabin depressurization.

3.3.1.2.2 Propulsion.

3.3.1.2.2.1 Main Propulsion Subsystem (MPS). The Orbiter vehicle main propulsion subsystem assisted by two liquid rocket boosters during the initial phase of the ascent trajectory shall provide the velocity increment and thrust vector control from lift-off to main engine shutdown with a maximum SSME gimbal deflection of ±11 degrees in pitch and ±9 degrees in yaw. *

3.3.1.2.2.2.1 OMS Burn Sequence. The OMS shall be capable of burning all of its allocated propellant in either a single long burn or a series of multiple burns spaced over the mission duration.

3.3.1.2.2.2.2 OMS Tank Sizing. The integral OMS pressurant/propellant tankage shall be sized for a delta V capability of (TBD) fps with a 65,000 pound payload.

Provisions shall be made to incorporate additional tankage capacity to achieve an overall propellant capacity of 2.5 times that of the integral tankage. The additional capacity shall be provided by supplementary propellant supply kits located in the payload bay clear volume and will be payload volume and weight chargeable items. The auxiliary tankage kits shall be designed such that either one, two, or three kits may be installed as required.

3.3.1.2.2.3 Reaction Control Subsystem (RCS). The RCS shall provide three-axis angular control and three-axis translation. The RCS shall provide translational delta V for Orbiter/ET separation, and rendezvous and docking as defined in 3.2.1.1.3. Vernier RCS thrusters shall be provided for use in angular control modes for low stability rates.

3.3.1.2.2.3.1 RCS Thrusters Installation. The RCS thrusters installation shall be such as to minimize angular crosscoupling during all RCS operations, and to minimize translational accelerations during rotational maneuvers, and rotational accelerations during translational maneuvers.

3.3.1.2.2.3.2 RCS Tank Sizing. The RCS tankage shall be sized to provide the RCS translational delta V requirements for accomplishing the missions specified in 3.2.1.1.3 and attitude control from main engine shutdown to initiation of OMS burn, the pointing accuracies specified in 3.3.1.1.1; and the attitude control on-orbit and during entry. Mission requirements that exceed the maximum feasible size of the RCS tanks may be accommodated by loading and using propellant in the OMS tanks.

The RCS tankage shall also have the capability to be off-loaded utilizing the PVT method, to a minimum 65% (by wt) of maximum rated loading for specific selected missions, as deemed necessary. *

3.3.1.2.3 Avionics.

3.3.1.2.3.1 General Requirements.

3.3.1.2.3.1.1 Auto landing. The avionics subsystems shall provide automatic landing capability (following acquisition of terminal area RF landing aid signals) through rollout for orbital missions with the Orbiter vehicle configured for non-propulsive atmospheric flight. The auto landing capability shall provide the control to:

a. Maintain the Orbiter on the proper glide slope during the landing approach phase,

b. Maintain the directional control necessary to bring the Orbiter to touchdown on the runway, 3.3.1.2.2.2 Orbital Maneuver Subsystem (OMS). An Orbital Maneuver System shall provide the propulsive thrust to perform orbit insertion, orbit circularization, orbit transfer, rendezvous, and deorbit.

c. Perform the final touchdown maneuver,

d. Maintain the Orbiter's heading on the runway in a safe orientation through rollout.

Rollout is defined as the portion of the landing from touchdown to the point where the Orbiter is brought to a full stop or can be manually turned off the runway onto a taxiway. The automatic landing capability shall incorporate rudder control, nose wheel steering, and/or differential braking to provide directional control. Deployment of the landing gear and application of the braking device shall be manual. Redundancy requirements shall be provided as specified in 3.3.1.2.3.1.2.

3.3.1.2.3.1.2 Redundancy. The avionics subsystem shall have sufficient redundancy capabilities, using the flight crew as necessary (with provisions for periods of simultaneous sleep), to provide mission completion after any single failure. After the minimum required time has elapsed to achieve the selected post-failure configuration by automatic or manual means (whichever is used for the given failure), the same subsystem shall have the ability to sustain any second failure and terminate the mission safely.

In addition to Criticality 1 and 2 single failure points, the items during intact abort not meeting the fail safe redundancy requirements shall be identified in the individual element critical items list.

The avionics subsystem shall not preclude the capability for abort specified in 3.2.1.5.1 herein.

Deviations/Waivers (TBD) *

3.3.1.2.3.1.3 COMSEC/TEMPEST Requirements for DOD. The TEMPEST requirements will be as specified in classified technical direction. An RF Orbiter TEMPEST Evaluation will be performed by a USAF team with NASA support.

3.3.1.2.3.2 Communications and Tracking Subsystem (C&T). The communications and tracking subsystem shall provide for:

a. Reception, transmission, and distribution of Orbiter, ground, EVA, and attached payload voice. EVA communications will be provided by Extravehicular Communicator (EVC). All external voice interfaces with the Orbiter shall be a nominal ODBM signal level, 300 to 3000 Hz pass band and a nominal 600 ohm balanced impedance.

b. Transmission of realtime and stored operational PCM data and LRB realtime operational instrumentation data, from launch to separation, as required for LRB flight evaluation and performance assessment. *

c. Reception of payload PCM telemetry in NRZ - L, M or S or Bi-Phase L, J, or S codes.

d. Transmission of payload commands in NRZ - L, M or S codes.

e. Receiving and decoding of ground-to-Orbiter commands.

f. Landing and atmospheric navigation RF aids, and on-orbit tracking to include, two-way doppler and GSRDN ranging.

g. Generation, transmission and distribution of television signals.

h. Tracking targets.

i. The installation and operations of GFE COMSEC equipment for encryption/decryption/authentication for DOD missions.

j. Transmission of main engine PCM data.

k. Reception of EVA data.

l. Reception and retransmission of payload data (including the handling of encrypted data and non-standard payload data) to the ground via either the S-Band PM downlink or the Ku-Band downlink operational data channel (selectable). Time correlation between Orbiter data and payload data shall be provided to within one millisecond.

m. Reception and processing of Global Positioning System (GPS) data to derive the current Space Shuttle navigation state. Use of GPS by the Space Shuttle, combined with GPC software for onboard deorbit targeting calculations, shall provide onboard navigation autonomy for the Space Shuttle. These capabilities have an operational effectivity of the fourth quarter of calendar year 1982.

The communication and tracking subsystem shall provide the capability to transmit and receive between the Orbiter and the following, subject to the compatibility requirements of the applicable Interface Control Document (ICDs).

a. Other space vehicles

b. Payloads

c. Extravehicular astronauts

d. Prelaunch checkout facilities

e. Air Traffic Control (ATC) facilities

f. Space Tracking and Data Network (STDN)

g. Tracking and Data Relay Satellite (TDRS)

h. Air Force Satellite Control Facility (AFSCF)

i. Orbiter vehicle landing site facilities

j. Ground navaids and facilities

The communications and tracking subsystem shall provide the capability of performing the following on-orbit functions simultaneously.

- Two-way phase coherent S-Band PM communication with either TDRS or a STDN ground station or an AFSCF ground station.

- S-Band FM transmission of TV or sideband data to either a STDN or an AFSCF ground station.

- Two-way S-Band communication with one detached NASA or USAF payload.

- The Ku-Band communication/rendezvous radar system has an effectivity of March 1981.

The S-Band PM links transfer single digital data streams which combine voice and telemetry for the downlink and combine voice and command for the uplink. The resulting Time Division Multiplexed (TDM) data streams shall be identical for the TDRS, STDN, and AFSCF links, although further processing shall be applied for transmission compatibility.

The Ku-Band forward link shall simultaneously transfer command data, two voice channels and wide-band data (for direct routing to the attached payload interface) to the Orbiter. The Ku-Band return link shall transmit data at maximum rate of 50 Mbps; the actual data rate chosen will depend on the Bit Error Rate (BER) requirements of the payload(s). The Effective Isotropic Radiated Power (EIRP) of the Orbiter Ku-Band return link shall be 48.8 dBW. The Ku-Band error rate between the Orbiter antenna and TDRSS ground station antenna shall be no greater than a one bit error in every 100,000 bits transmitted (minimum BER of 1×10^{-5}). Ku-Band shall have the capability to relay to the ground (in a bent-pipe mode) data from attached or detached payloads.

The Ku-Band return link shall operate in either of two selectable modes; one offering simultaneous transmission of three channels.

Utilization of the various channels will be on a time-shared basis. The informtion to be transmitted via Ku-Band shall consist of various combinations of the following signals:

a. Realtime operational data

b. Uplink text and graphics data (144 Kbps forward link)

c. Recorder dumps

d. Wideband digital data (up to 50 mbps max.)

e. Payload Standard TV

f. Orbiter TV

g. Wideband analog data (including non-standard TV)

3.3.1.2.3.2.1 S-Band Antenna. The Orbiter S-Band subsystem shall have a minimum antenna gain of +4 dB over a spherical coverage of at least 85%.

3.3.1.2.3.3 Guidance, Navigation, and Control Subsystem. The GN&C subsystem in conjunction with supporting subsystems shall be capable of providing guidance, navigation, and control for the flight vehicle through all phases of flight from launch through landing and for aircraft aerodynamic flight modes. The control system shall have capability to provide Modal Suppression and/or attenuation as required for dynamic stability. Modal Suppression for control of dynamic loads (if necessary) shall be accomplished within the constraints of hardware and stability requirements.

3.3.1.2.3.3.1 GN&C Ground Support. The GN&C subsystem shall provide the capability for processing uplinked and downlinked parameters as specified in NSTS 07700, Volume XVIII.

3.3.1.2.3.3.2 Manual Control. A digitally processed manual control capability shall be provided for all flight control functions for all Orbiter alone flight phases. During launch and ascent of the mated vehicle (ORB/ET/LRB and ORB/ET) the capability to digitally process manual main engine and LRB engine throttle commands shall be provided. Manual throttle and manual guidance (autoguidance command incremented or replaced by stick command) capability shall be provided during mated vehicle flight phases for contingency situations. If manual guidance is selected, there is no requirement for return to automatic operation during ascent. Provision for integrated SSME/LRB engine manual throttle control shall be provided. *

3.3.1.2.3.4 Display and Control Subsystem. The displays and controls subsystem shall provide the crew with the following basic capabilities during all normal and contingency operations: (a) the means to monitor and command vehicle rotation, translation, and flight path; (b) the means to monitor and command onboard subsystems; (c) the means to monitor and command critical attached payload functions; and (d) the means to detect and safe hazardous conditions. In addition, the D&C shall provide all crew compartment interior and integral lighting.

3.3.1.2.3.4.1 Abort Commands. The Orbiter vehicle shall be the command center for all abort commands. The Orbiter shall monitor critical vehicle subsystems to identify failures, generate automatic and/or manual abort signals, display abort signals, display abort conditions, and control automatic abort initiations commands.

3.3.1.2.3.5 Data Processing and Software Subsystem. The airborne data processing and software subsystem shall provide computational capabilities for guidance, navigation and control; subsystems performance monitoring and display; payload checkout, monitoring, caution and warning, display, discrete commanding and command loading; and the capability to work in conjunction with the ground system for performing ground functions.

3.3.1.2.3.6 Electrical Power Distribution and Control (EPDC) Subsystem. The EPDC subsystem shall provide conditioning, conversion, control, distribution of electrical power supplied by the electrical power generation subsystem. The EPDC subsystem shall also provide all Orbiter vehicle lighting external to the crew compartment.

3.3.1.2.3.7 Instrumentation Subsystem. TBD *

3.3.1.2.3.8 Performance Monitor. A performance monitor function shall be provided utilizing elements of the instrumentation, display and control, and data processing and software subsystems. This function shall provide to the flight crew information concerning health status, configuration status, and fault detection and isolation status for flight vehicle subsystems. This function shall also support redundancy management to the level required in flight; onboard fault detection, isolation and anomaly recording; management of Orbiter data recording; and monitoring and management of certain other inflight functions. An interface shall be provided for use of the onboard capabilities in support of ground operations.

3.3.1.2.3.9 Closed Circuit TV Subsystem. A closed circuit TV subsystem shall be provided that consists of a Video Control Unit, VCU, (remote control unit plus video switching unit), two cabin monochrome CCTV monitors, and wiring, mounts, and controls to support the following TV camera services:

a. Two color cameras in cabin.

b. Two cameras for payload bay bulkheads, with pan and tilt and onboard remote control functions (one camera on each payload bay bulkhead).

c. One camera on each RMS arm, with pan and tilt and remote control function capability for the forearm location.

d. Payload bay-mounted camera(s) to aid in "x" coordinate alignment when using the RMS. Capability shall also include EVA portable TV coverage.

The Video Control Unit (VCU) shall support split screen capability on the CCTV monitors plus other locations including payload and downlink. The VCU shall also support CCTV uplink command capability.

3.3.1.2.3.10 Biomedical Monitoring. Capability shall be provided such that any crew position, including payload specialist, can be monitored for ECT activity during the launch and entry phases of the Orbital Flight Tests. Capability will also be provided for prelaunch realtime acquisition of ECG data from the Commander and Pilot. After completion of the Orbital Flight Tests, ECG monitoring capability will be limited to the Payload Specialists positions.

3.3.1.2.3.11 Uplink Text and Graphics Subsystem. The capability shall exist in the Orbiter to receive text and graphics data which has been transmitted from the Mission Control Center - Houston (MCC-H) via the Tracking and Data Relay Satellite System (TDRSS) and produce a hardcopy for use by the flight crew. The uplink text and graphics subsystem shall receive data from the Ku-Band subsystem. Text and graphics data shall comprise 128 Kbps of the Ku-Band 216 Kbps uplink. The 128 Kbps data stream shall be transmitted to the Orbiter/Spacelab interface as well as to the Orbiter uplink text and graphics hardcopier. The uplink text and graphics subsystem shall satisfy the following resolution and grey level requirements:

Mode 1 - 125 lines/inches resolution and 2 linear grey levels

Mode 2 - 250 lines/inch resolution and 64 linear grey levels

Mode 3 - 350 lines/inch resolution and 2 linear grey levels

Mode 4 - 350 lines/inch resolution and 64 linear grey levels

An interim teleprinter system will be utilized until such time as TDRSS and the uplink text and graphics subsystem is operational. This interim teleprinter system utilizes the S-Band uplink voice channel #2 on a time share basis and an appropriate interface device and hardcopy printer on the Orbiter.

3.3.1.2.4 Environmental Control and Life Support Subsystem (ECLSS). The ECLSS shall provide the life support for the flight personnel as specified in 3.3.1.2.1.1 and environmental control for the Orbiter vehicle during all mission phases. The ECLSS shall provide the life support environment required to provide a shirtsleeve environment for the crew. The ECLSS shall perform the major functions of a atmosphere revitalization, active thermal control, water, waste and food management, smoke detection, and fire suppression within pressurized cabin and avionics bays. Provisions shall also be made for support to extravehicular/intravehicular activity (EVA/IVA) and a GFE atmospheric trace gas analyzer.

3.3.1.2.4.1 Crew Compartment Atmosphere.

3.3.1.2.4.1.1 Total Pressure. The total pressure shall be 14.7 ± 0.2 psia using a two-gas system composed of nitrogen and oxygen. The Orbiter shall have the capability during on-orbit EVA operations to operate in either of two modes:

a. Reduced cabin pressure procedure - the total pressure control range shall be 10.2 +/- 0.2 psia with caution and warning limits set at no lower than 10.0 and no higher than 10.6 psia. The % oxygen shall not exceed 30% (including sensor errors). The Pressure Control System (PCS) shall be manually controlled during this procedure.

b. Insuit Prebreath Procedure - the total pressure shall be maintained at 14.7 +/- 0.2 psia except during airlock repressurization, when the total pressure shall be allowed to drop to 13.7 psia minimum. The total time for this pressure excursion below 14.5 psia shall be limited to 30 minutes.

NOTE: All payload hardware located in the orbiter crew cabin shall be certified safe in all of the above environments.

3.3.1.2.4.1.2 Oxygen Partial Pressure. The partial pressure of oxygen shall be 3.2 $\pm$ 0.25 psia at total pressure of 14.7 +/- 0.2 psia. The orbiter shall have the capability during on-orbit EVA operations to operate in either of two modes:

a. Reduced Cabin Pressure Procedure - The oxygen partial pressure control range shall be from 2.55 to 2.8 psia and shall be constrained by caution and warning limits set at no lower than 2.55 and no higher than 2.9 psia at a total pressure of 10.2 + .4, - .2 psia. The % oxygen shall not exceed 30% (including sensor errors). The Pressure Control System (PCS) shall be manually controlled during this procedure.

b. Insuit Prebreath Procedure - The partial pressure of oxygen shall be maintained at 3.2 +/- 0.25 psia except during airlock repressurization, when the oxygen partial pressure shall be allowed to drop to 2.7 psia minimum. The total time for this pressure excursion below 2.95 psia shall be limited to 30 minutes.

NOTE: All payload hardware located in the orbiter crew cabin shall be certified safe in all of the above environments.

3.3.1.2.4.1.3 Carbon Dioxide Partial Pressure. The carbon dioxide partial pressure shall be:

Nominal: 5.0 mmHg
Range: 0 - 7.6 mmHg

3.3.1.2.4.1.4 Cabin Temperature. The cabin air temperature shall be 65 - 80F during all mission modes except entry to egress (assuming 15 minutes maximum after touchdown) when it shall not exceed 90F.

3.3.1.2.4.2 Crew Exposure (Max Temperature). The crewmen shall not be exposed to direct contact temperatures greater than 113^{o}F on equipment or structure normally touched. This excludes such items as windows and certain structure during entry through rollout, that are accessible to the crew, but should not normally be touched.

3.3.1.2.4.3 ECLSS for Ferry. Provisions shall be made for the necessary ECLSS functions to support ferry operations.

3.3.1.2.4.4 Emergency Conditions. Provisions shall be made for the following emergency conditions:

a. Cabin emergency repressurization from space vacuum

b. Cabin emergency pressure maintenance

c. In-orbit survival

d. Emergency bailout mode.

The baseline contingency expendables shall be sufficient to support the worst of any one of these contingencies (i.e., contingencies are not additive). Additional expendables necessitated by other mission options (increased crew size) shall be provided as a payload penalty.

3.3.1.2.4.4.1 Cabin Emergency Repressurization from Space Vacuum. Provisions shall be made for one cabin repressurization in the event of an emergency which required depressurization of the cabin to facilitate crew rescue. The flowrate shall be such as to repressurize the cabin in approximately 1 hour. The ECLSS baseline does not include a dedicated means of crew life support or support of other cabin located subsystems equipment while the cabin is depressurized.

3.3.1.2.4.4.2 Cabin Emergency Pressure Maintenance. Provisions shall be made for maintaining a cabin pressure of 8.0 ± 2.0 psia with an oxygen partial pressure of 2.2 ± 0.25 psia, a flowrate equivalent to the leakage of a 0.45 inch diameter hole for a return time, from orbit, of 165 minutes. Expendables provisioning shall be sufficient to support a maximum crew of seven during this contingency. System design shall accommodate a crew up to 10.

3.3.1.2.4.4.3 Crew Survival In-Orbit. The Orbiter vehicle shall have the capability to support the survival of a four-man crew for 96 hours after an in-orbit contingency, assuming reduced consumption rates as appropriate, and with the crew in a resting level of activity and the vehicle essentially powered down.

3.3.1.2.4.4.4. Emergency Bailout Mode. Provisions shall be made for all crew members to safely escape from an orbiter during controlled subsonic gliding flight conditions. Protective survival equipment shall be provided to sustain the crew below 70,000 feet altitude and for 24 hours after a water landing.

3.3.1.2.4.5 Waste Management. The waste management, personnel hygiene facility shall be a permanent installation in the Orbiter utilizing the same equipment for all missions and shall accommodate both male and female personnel. Elements of the waste management system may be removable for cleaning. All solid waste shall be stored for return to earth for the seven day (42 man-days) design mission. Urine, condensate and personal hygiene waste water shall be stored in one waste tank with managed waste water dumps to reduce experiment contamination and to provide emergency flash evaporator capabilities for planned descent contingencies.

3.3.1.2.4.6 EVA/IVA Operations Support. Two Extravehicular Mobility Units (EMU) (composed of space suits and life support systems) and a maximum of two Personnel Rescue Systems (PRS) to support unscheduled or contingency EVA/IVA operations shall be provided for all missions at the expense of personnel group weight. EMUs and PRSs in excess of the foregoing requirements shall be at the expense of payload weight ahd volume as specified in Paragraph 3.2.1.1.2.1.3. Applicable design requirements for EVA/IVA operations support are as follows:

a. An EVA Service and Recharge Station to support, recharge, checkout, and for donning of EVA equipment shall be provided by the Orbiter. Weight and volume are chargeable to the Orbiter.

b. ECLSS expendables shall be provided by the Orbiter for:

1. Three airlock pressurizations except when the Spacelab tunnel adapter is used in series with the airlock for EVA; then two airlock/tunnel adapter pressurizations shall be required.

2. Six one-man prebreath operations.

3. Three two-man EVA equipment recharges. (Equipment may be precharged before installation to provide for a total of three, two-man EVAs). Expendables for one, two-man EVA, operations must be reserved for any contingency Orbiter operations: i.e., external inspection, repair or rescue as required for safe return of crews. The time duration for each EVA/IVA shall be six hour maximum. System Design shall allow for .5 hour for egress and ingress and .5 hour reserve in addition to the six hour EVA.

4. Expendables to support a Manned Maneuvering Unit shall be charged to payload when used in support of planned EVA for payloads.

c. The configuration of the EMU shall permit passage through the Orbiter interdeck access hatch.

3.3.1.2.4.7 Emergency Oxygen. Provisions shall be made for connecting oxygen mask assemblies for EVA/IVA oxygen prebreathing and emergency conditions.

3.3.1.2.4.8 Water Management. The water management system shall be a permanent installation in the Orbiter (four supply tanks) which provides potable water for drinking, food preparation, EVA recharge and for heat rejection evaporant. The water management system shall have a minimum storage management capability of 12 hours and 18 kw power level of fuel cell generated water between overboard nozzle dumps in orbit. Excess water shall be periodically dumped, via direct dump, through heated nozzles or with weight chargeable to the payload per Paragraph 3.2.1.1.2.1.12.

3.3.2.1.5 Power.

3.3.1.2.5.1 Electrical Power Subsystem. The electrical power subsystem shall generate the electrical power required for all Orbiter vehicle subsystems. It shall also satisfy power requirements for:

a. The full duration 7-day orbital mission

<u>Note</u>: The inert weight associated with the 1530 KwH power system shall be chargeable to the Orbiter. The inert weight for an additional reactant storage kit (840 KwH) plus the cryo consumables for both shall be chargeable to Level II as specified in NSTS 0700 Volume X, Appendix 10.12. *

b. The approach and landing Test Flights

c. Emergency restart or reset of the prime power components.

d. Four-day survival period after an on-orbit contingency that occurs at the end of the Design Reference Mission. This contingency shall be satisfied with no Power Reactant Supply and Distribution (PRSD) subsystem failures.

e. The ET and LRB from prelaunch to separation. *

f. Orbiter landing station power until connection to the fixed facility power supply. Usage of the contingency cryo reserve, as available, shall satisfy this requirement. In the event of contingency cryo depletion, contingency ground power will be provided by the Orbiter landing station.

3.3.1.2.5.1.1 Electrical Power Subsystem Cryo Loading. The Shuttle System shall be capable of off-loading electrical power subsystem cryogenic reactants for specific selected missions, as deemed necessary. *

3.3.1.2.5.2 Hydraulic Subsystem. The hydraulic subsystem, consisting of hydraulic pumps, actuators, fluid distribution lines, and heating and cooling provisions, shall provide power to all hydraulic users.

3.3.1.2.5.2.1 Hydraulic Power. Hydraulic power, provided by the APU subsystem shall satisfy power requirements for:

a. The full duration 7-day Orbital mission

b. The approach and landing test flights

c. Integrated system checkout and prelaunch activity

d. Post-Landing activities as required

e. 30 day missions

3.3.1.2.5.2.2 Hydraulic Design. Hydraulic subsystem design and installation, shall be in accordance with MIL-H-5440. This specification (MIL-H-5440) shall take precedence over safety factors stated in Paragraph 3.2.2.1.5.2.

3.3.2.1.5.3 Auxiliary Power Unit Subsystem. The Auxiliary Power Unit (APU) subsystem consisting of APUs, APU controllers, fuel tanks, fuel distribution system, exhaust ducts, thermal control system, shall provide power necessary for the hydraulic subsystem.

3.3.1.2.5.3.1 Auxiliary Power Unit Subsystem Propellant Loading. The Shuttle System shall be capable of off-loading APU subsystem propellant for specific selected missions as necessary.

3.3.1.2.6 Crew Provisions.

3.3.1.2.6.1 Emergency Egress. The Shuttle System shall provide for emergency egress of the crew and passengers in the vertical mode on the launch pad within a total of 2 minutes; 30 seconds to egress arm and 90 seconds to a secure area, without action by ground personnel with access arm in egress position. The

Orbiter vehicle shall incorporate onboard provisions to place the Orbiter in a safe condition following landing and permit unaided crew egress. Provisions shall be made for emergency egress of the crew after landing rollout within 60 seconds.

3.3.1.2.6.2 Crew and Cargo Transfer. The Orbiter vehicle shall provide shirtsleeve access to pressurized payload modules and direct pressure suit access (via airlock) to the unpressurized payload bay in flight. The Orbiter shall provide handholes and handrails to allow crewmen to translate from EVA exit(s) to EVA work areas. Also the Orbiter will provide attach points for tethers, umbilicals and other EVA aids.

3.3.1.2.6.3 Normal Ingress/Egress. The Orbiter cabin arrangement shall provide for crew and passenger ingress and egress with the Orbiter in the vertical position on the launch pad during normal operations. With the Orbiter in the horizontal position, the cabin arrangement shall provide for normal crew and passenger ingress and unaided egress via ground supplied equipment.

3.3.1.2.6.4 (Deleted).

3.3.1.2.7 Cabin Arrangement.

3.3.1.2.7.1 Flight Station. A flight station shall be provided for the commander and pilot.

3.3.1.2.7.1.1 Single Crewman Control. The flight station shall be arranged to allow a crewman, flying from either seat, to return the Orbiter vehicle to earth. For operational flights the design shall allow for the elimination of "single crewman control" capability from one of the seats.

3.3.1.2.7.2 Airlock. (Deleted).

3.3.1.2.7.3 On-Orbit Station. An on-orbit station shall be provided to accommodate attitude and translation control of the Orbiter while maintaining direct visual viewing of other orbital elements. The on-orbit station shall also accommodate docking control and operation of the remote manipulator system and provide for simultaneous direct and/or remove viewing (CCTV) of the manipulator payload handling, and payload experiment operations. The on-orbit station shall also provide the following:

a. Capability for controlling of payload bay doors with sufficient direct visibility of the doors to ensure proper operation.

b. Capability to support dual (side-by-side) operators.

c. Capability for some dedicated payload DC space and payload-associated equipment volume, plus capability to support mission/payload functions by accommodating ground changeout of mission-unique equipment (such as manipulator DC) to payload-unique equipment.

3.3.1.2.7.4 Mission Station. A mission station shall be provided to accommodate monitoring and managing of selected Orbiter systems, and monitoring managing and sending commands to attached and detached payload support systems, and conducting some payload operations. The mission station shall also provide the following:

a. Standard mission capabilities to support Orbiter and payload operations, such as communications, electrical power and consumables management.

b. Monitor critical functions (including C&W) of attached payloads and issue of appropriate safing commands.

c. On-orbit work surface area, which includes some temporary on-orbit stowage capability.

3.3.1.2.7.5 Payload Station. A payload station shall be provided to accommodate management of payload operations. The payload station shall also provide the following capabilities:

a. Interfaces necessary for supporting payload supplied D&C and equipment, including an encapsulated DOD CIU.

b. Capability to support single operator and restricted dual (side-by-side) operators.

c. On-orbit work surface area, which includes some temporary on-orbit stowage capability.

3.3.1.2.7.6 Changeout of Aft Crew Station equipment. The cabin aft crew stations (mission station, payload station and on-orbit station) will be designed as appropriate to facilitate changeout and installation of Orbiter and payload supplied display and control panels and equipment. Equipment beyond the standard crew stations provisions shall be charged to payload weight.

3.3.1.2.7.7 Photographic Stations. Photographic stations shall be provided with provisions for electrical power and structural attach points for the GFE camera bracket. Photographic stations shall be located in the following areas with the indicated capabilities:

a. Flight deck pilot station - Attach points for forward looking camera and crew observation cameras. Attach points should support cameras during dynamic mission phases, i.e., launch and reentry.

b. Flight deck aft crew station - Attach points for overhead and cargo bay window, to support on-orbit camera installation.

c. Mid-deck general interior - Attach points for documenting crew habitation at 2 locations. These stations support on-orbit camera installation.

d. High Optical Quality Scientific Photo Station - A high optical quality window shall be provided in the side hatch of the crew module.

3.3.1.3 Orbiter Interface Characteristics.

3.3.1.3.1 Orbiter Interface with External Tank. The Orbiter vehicle shall interface with the ET as defined in the Orbiter Vehicle/External Tank ICD 2-12001.

3.3.1.3.1.1 Orbiter/External Tank Release. The Orbiter vehicle shall provide Orbiter/external tank attachment and release. The release mechanism shall be retained with the Orbiter.

3.3.1.3.1.2 Orbiter/ET Umbilical.

3.3.1.3.1.2.1 Conductive Signal Path Umbilical. A conductive signal path umbilical shall be provided between the Orbiter vehicle and external tank for the following functions:

a. ET Status Monitor

b. Sequence Commands to ET

3.3.1.3.1.2.2 Fluid Umbilicals. Fluid umbilicals shall be provided between the Orbiter vehicle and external tank for the following Main Propulsion Subsystem (MPS) interface functions:

a. LO_2 and LH_2 feed, fill and drain

b. LH_2 recirculation

c. Pressurization for LH_2 tank

d. Pressurization for LO_2 tank

3.3.1.3.1.3 Interface Access. Routine ground service operations at the Orbiter/ET interface shall not be required after rollout.

3.3.1.3.2 Orbiter Interface with Liquid Rocket Booster. The Orbiter/LRB functional interface requirements are (TBD) *

3.3.1.3.2.1 Ascent Guidance and Control. All navigation and guidance functions for the mated flight configuration shall be performed by the Orbiter vehicle. * Flight control shall be performed jointly by the Orbiter vehicle and the LRBs from lift-off to staging. The Orbiter shall provide GN&C commands to the LRBs. Steering commands to LRB TVC shall be provided by the LRB on-board avionics.

3.3.1.3.2.2 LRB Power Bus Redundancy. No single failure of an active * component on the Orbiter shall result in the permanent loss of a LRB power bus.

3.3.1.3.3 Orbiter Interface with Payload. The Orbiter interfaces with payloads shall be defined in accordance with the provisions of NSTS 07700, Volume XIV, and ICDs 2-05101, 2-05201, and 2-05301.

3.3.1.3.3.1 Payload Carriers. The Orbiter vehicle shall interface with a series of standard GFE payload carriers, such as the spacelab/airlocks/pallets, mounting platforms, propulsions systems, and free flying systems. Standard carriers shall be designed to be compatible with the Orbiter vehicle.

3.3.1.3.3.1.1 Payload Structural Attachment. The Orbiter vehicle structure shall provide multiple sets of mounting points for a statically determinate structural attachment subsystem. Statically indeterminate payload attachment schemes shall not be precluded, but such schemes must be compatible with the structural and mechanical capability of the Orbiter attach points for all combinations of deflections and loads.

3.3.1.3.3.2 Fluid System Interfaces. The Orbiter shall be capable of accommodating the prelaunch servicing, inflight fluid functions, and post-landing deservicing of payload fluids in a manner consistent with the safety of the crew, ground personnel, and the Orbiter. These accommodations shall provide for the required intact abort capability as well as for normal missions. In order to provide operational flexibility and selection of best operations for any mission, the prelaunch servicing functions shall be achievable by preinstallation loading, by loading on the launch pad through the open payload bay doors, and by loading on the launch pad through Orbiter lines with the bay doors closed. The payload fluid systems shall be serviced before installing the payload in the Orbiter whenever possible. Dump in flight of payload fluids shall be provided when required (e.g., Orbiter CG control, reduced landing weight, avoidance of payload structural weight penalty, or safety). Flight equipment weights for any permanently installed fill, vent, pressure relief, and drain system hardware, and dump hardware shall be chargeable to the Orbiter. Weight of flight equipment kits required for dump of payload fluids and for payload repressurization shall be chargeable to the payload.

3.3.1.3.3.3 Electrical Interfaces.

3.3.1.3.3.3.1 Electrical Power Interfaces. The Orbiter/payload power transfer circuits shall have power handling capabilities as listed below. More than one feeder may be used simultaneously by the payload to receive power, but power feeders from separate Orbiter sources will not be tied together directly by the payload. The two auxiliary feeders at the mid payload bay, however shall be so mechanized that they may be tied together in the payload without additional circuitry in the payload.

a. Mid Payload Bay Power Interface - One interface at main DC distribution assembly No. 3 shall be capable of delivering the entire rated output (12kw) from fuel cell No. 3 or 8 kw from Main DE Bus B. This power shall be transferred to an interface near station Xo = 693 on the starboard side of the payload bay by a harness, which is part of the Orbiter and Orbiter vehicle weight chargeable. Provisions shall be made to allow the installation of a kit to route the 8 kw from Main DC Bus B directly to another interface near station Xo = 693. If this kit, which is not now baselined, is installed the power from Main DC Bus B will no longer be available through the 12 kw feeder. This kit, if added, will be payload weight chargeable.

b. Auxiliary Power Interface - Two feeders capable of being tied together directly by the payload shall be provided on separate connectors near the power interface at station Xo = 693. Each feeder shall be capable of providing 20 amps from separate Orbiter sources, and the two feeders, when connected in parallel, shall have a total capacity of 20 amps.

c. Aft Payload Bay Interface - Two 2 kw feeders shall provide power from separate Orbiter sources near Station Xo = 1307, one on each side of the bay. In addition ground power will be available thru the right hand T-0/Xo 1307 bulkhead payload interface panel.

3.3.1.3.3.3.1.1 Aft Crew Station Payload Unique Equipment. Redundant Orbiter sources shall provide electrical power to payload equipment in the aft crew station.

3.3.1.3.3.3.1.2 Orbiter Emergency Minimum Power Conditions. Under Orbiter emergency minimum power conditions, main and auxiliary power will be terminated to payload and Orbiter aft flight deck payload equipment after payload safing. If required, a minimum power level (up to 200 watts) will be sustained to keep the payload safe through entry, landing, and removal.

3.3.1.3.3.3.2 Electrical Signal Interface. The Orbiter vehicle shall provide the electrical signal wiring interface at the forward cargo bay bulkhead to accommodate payload supplied mission equipment mounted in the Orbiter cabin; primary command, systems management and telemetry; caution and warning; guidance, navigation and control; timing; data recording; audio; closed circuit TV; S-Band FM; and Ku-Band. A patch distributor shall be located in the Orbiter aft cabin to provide flexibility for mission unique equipments and minimize wiring changes during turnaround. The Orbiter shall provide a signal wire interface between the aft bay bulkhead and the T-0 umbilical which accommodated command, telemetry, safing, caution and warning and status data from the payload independent of the Orbiter avionics subsystem. In addition, the Orbiter shall make provisions for routing payload signal wire harnesses through the central portion of the payload bay wire trays (port or starboard) with provision for removal and installation independent of Orbiter wiring and without uncovering the Orbiter wiring. The above provisions are Orbiter weight chargeable. Kits required to interface the payload with the forward and aft cargo bay bulkheads are payload provided and weight chargeable.

3.3.1.3.3.3.3 Power Allocation. The Orbiter shall allocate power to the payload as stated below, but usage is constrained by heat removal capacity as specified, in Paragraph 3.3.1.3.3.6.1.

3.3.1.3.3.3.3.1 Ground Operation (GSE Power). The Orbiter shall provide power through the mid payload bay power interface for use by the payload during checkout and prelaunch at the levels and times noted below:

a. Before the Orbiter is transferred to internal power it shall provide 3 kw of GSE power to the payload from an Orbiter main bus which is supplying as much as 9 kw to Orbiter loads. Both the primary and backup payload power circuits shall be capable of supporting this requirement. Reference NSTS 07700, Volume XIV, Attachment 1 (ICD 2-19001) Table 7.2.1-1.

b. The Orbiter shall provide on-orbit power levels (approximately 12 kw peak) through the primary power circuit or contingency power levels (8 kw peak) through the backup power circuit. This power shall be time-shared with the Orbiter.

3.3.1.3.3.3.3.2 Prelaunch (Internal Power Source), Ascent, Descent, and Post Landing (Internal Power Source). The Orbiter shall provide 350 watts average, 420 watts peak to the Aft Flight Deck for payload unique operation functions equipment, and a total of 1000 watts average, 1500 watts peak to the Orbiter/payload bay electrical interface. Peaks limited to a maximum of 2 minutes per phase.

3.3.1.3.3.3.3.3 On-Orbit. The Orbiter shall provide 750 watts average, 1000 watts peak to the Aft Flight Deck for payload unique operation functions equipment and

a. 7000 watts maximum except for peaks up to 12,000 watts (for a maximum of 15 minutes to occur no more often than once in a 3.0 hour period) at the mid payload bay electrical interface, or,

b. 1500 watts average, 2000 watts peak from each feeder at the aft payload bay electrical interface.

3.3.1.3.3.3.3.4 Electrical Energy. The Orbiter EPS shall provide 50 KwH of DC electrical energy to the payload. To support payloads requiring more than 50 KwH provisions shall be made by the Orbiter for installing one reactant storage kit outside the payload envelope. Volume for three additional kits shall be provided outside the paylcad envelope. Each kit shall be capable of providing approximately 840 KwH of additional energy. The weight of the kits, including reactants, shall be charged to the payload.

3.3.1.3.3.3.3.5 Operating Voltage and Ripple Voltage. Power shall be provided to the payload from a nominal 28 volt DC system.

a. Aft Flight Deck: 24.0 to 32.0 volts DC with peak-to-peak narrowband (30 Hz to 7 Hz) not to exceed 0.9 volts falling 10 dB per decade to 0.28 volts peak-to-peak at 70 kHz, thereafter remaining constant to 400 MHz. The momentary coincidence of 2 or more signals at any one frequency shall not exceed the envelope defined as 1.6 volts peak-to-peak (30 Hz to 7 kHz), falling 10 dB per decade to 0.5 volts peak-to-peak at 70 kHz, thereafter remaining constant to 400 MHz.

b. Mid Payload Bay: 27.0 to 32.0 volts DC with peak-to-peak narrowband (30 Hz to 7 kHz) not to exceed 0.9 volts falling 10 dB per decade to 0.28 volts peak-to-peak at 70 kHz, thereafter remaining constant to 400 MHz. The momentary coincidence of 2 or more signals at any one frequency shall not exceed the envelope defined as 1.6 volts peak-to-peak (30 Hz to 7 kHz) falling 10 dB per decade to 0.5 volts peak-to-peak at 70 kHz, thereafter remaining constant to 400 MHz.

c. Aft Payload Bay - Same as that of the Aft Flight Deck.

3.3.1.3.3.3.3.6 AC Power to Payloads. The Orbiter shall provide redundant AC power to the payload and mission stations in the aft flight deck at 400 Hz, 155 ± volts, 3 Phase.

a. On-orbit: 690 VA max.

b. Descent and Post-Landing: 350 VA max. continuous, 420 VA peak

Total combined DC and AC power to the aft flight deck shall not exceed the values specified in Paragraph 3.3.1.3.3.3.3.3.

3.3.1.3.3.4 Power Transition.

a. After Ascent: The Orbiter shall provide the capability for the payload to begin the transition from ascent power level to the on-orbit power level at the OMS-2 burn time plus 30 minutes. (The OMS-2 maneuver time is dependent upon orbital parameters and will vary between 34 and 57 minutes after launch.) The payload power increase may then proceed at a rate determined by the normal power-up procedures applicable for specific payload if the Orbiter has been appropriately powered down.

b. Prior to Deorbit: The transition from on-orbit power levels to entry power levels shall occur at the beginning of deorbit preparation (depending upon payload bay door closure).

3.3.1.3.3.4 Communications Interfaces.

3.3.1.3.3.4.1 Voice. The Orbiter shall provide a voice distribution subsystem with Orbiter/attached payload and ground/orbiter/attached payload duplex voice service, including conference capability with an attached payload voice service shall be provided at the mission specialist station.

3.3.1.3.3.4.2 Commands and Update. The orbiter vehicle shall have the capability to initiate and transmit up to 2 Kbps (information rate) of commands or data to an attached or released payload. The Orbiter shall provide the capability to issue commands via the Payload Signal Processor (PSP) to up to five attached payloads. In addition, the capability to command one detached payload shall be provided during that period of time that payload bay doors are open. This communication link shall include a command confirmation capability. The Orbiter shall have the capability to relay up to 2 Kbps ground initiated commands or data to attached payloads.

3.3.1.3.3.4.3 Digital data. Digital data shall be transferred from the payload to the ground via the orbiter vehicle as follows:

a. Attached payload - Up to 73.6 Kbps to be shared by all payloads. Each of up to 5 payloads shall provide a single time division multiplexed data stream to the Orbiter, not to exceed 64 Kbps.

b. Released payload - Up to 16 Kbps of digital data, including command confirmation, shall be relayed to the ground via Orbiter.

3.3.1.3.3.4.4 Television Video and Wideband Data. A hardwired input to the Orbiter vehicle wideband transmitter carrier shall be provided for attached payloads. For analog data, the payload shall provide commutation and subcarrier oscillators compatible with the Orbiter transmitter circuitry. For digital data, the payload shall provide the required encoding for compatibility with the Orbiter transmitter. This transmitter shall be time shared among Orbiter downlink television, payload analog data, or payload digital data.

3.3.1.3.3.4.5 Encryption requirements. Provisions to provide encryption/decryption/authentication for DOD data to be exchanged by RF between DOD spacecraft and the Shuttle flight vehicle will be internal to the DOD provided Communications Interface Unit (CIU). A single switch shall be provided to "zeroize" all encryption/decryption devices (KGX-60 plus KGT-60 and KGR-60 in the Payload station) during emergency situations.

3.3.1.3.3.4.6 Standardized Communications Interface. A standardized interface shall be provided by the Orbiter vehicle for communications between the Orbiter vehicle and payloads. The following functions shall be accommodated by standardized hardwire interfaces for attached payloads (1) wideband informatioin (analog or digital data, or television video) to the wideband transmission system; (2) digital data to the Orbiter telemetry system and (3) duplex voice to the Orbiter audio system. The following functions, as required, shall be accommodated by standardized RF interfaces for released payloads, (1) command or data transmission to the payload; (2) data transmission from the payload to the Orbiter; and (3) tracking of the payload.

3.3.1.3.3.4.7 Non-Standardized Communications Interface. A non-standardized interface shall be accommodated by the Orbiter to relay to the ground (in a bent-pipe mode) data from attached payloads.

3.3.1.3.3.5 Support Requirements.

3.3.1.3.3.5.1 Payload Monitor Subsystem Interface. Connectors shall provide a serial, digital data interface for payload performance monitoring and predeployment checkout. If command/stimulus functions are required to perform predeployment checkout, the payload system shall provide this capability via a serial, digital command link addressable by the payload monitor subsystem.

3.3.1.3.3.5.2 GN&C Data Interfaces. An interface between the payload and Orbiter vehicle GN&C shall be capable of providing transfer of payload initializing data, such as, vehicle state vector, attitude, and attitude rate. In addition, Greenwich Mean Time (GMT), mission elapsed time and other synchronization data shall be available from the Master Timing Unit (MTU). The capability shall be provided for transfer to the Orbiter GN&C computer of payload mounted sensor attitude information necessary to meet pointing accuracy and stability requirements specified in 3.3.1.1.1. Capability shall be provided to accomplish this transfer automatically via a hardwire interface and manually by the Orbiter crew as appropriate to the requirements of a specific payload.

3.3.1.3.3.5.3 Displays and Controls Interfaces. The Orbiter shall provide the following displays and controls to support payload operations:

3.3.1.3.3.5.3.1 On-Orbit Station. The On-Orbit Station shall contain D&C required to execute attitude/translation and maneuver sequences for rendezvous and docking, and deploy and retrieve payloads. This station shall also provide space and installation provisions for payload supplied equipment for conduct of payload operations.

3.3.1.3.3.5.3.2 Payload Station. The Payload Station shall contain space and installation provisions for payload supplied displays and controls which are used for management of payload operations. Payload Station displays and controls will be provided by, and charged to, payloads.

3.3.1.3.3.5.3.3 Mission Station. The Mission Station shall contain displays and controls for management of orbiter/payload interfaces, and shall provide accommodations for payload supplied displays and controls for payload support systems and for experiment operations as appropriate.

3.3.1.3.3.5.3.4 Payload Operations Monitoring and Control. The Orbiter shall provide the capability to support control of payload operations simultaneously with direct and/or remote viewing of selected payload components in the payload bay and/or the remote manipulator and its attachment points with payloads in the vicinity of the payload bay. The Orbiter shall provide the capability for simultaneous direct payload bay viewing and control of CCTV and appropriate interior/exterior lighting.

3.3.1.3.3.5.4 Orbiter/Payload Electrical Interfaces. The following electrical interfaces shall be provided between the Orbiter crew compartment and payloads:

a. Connectors to interface Orbiter avionics subsystem and payload supplied equipments to the payloads;

b. Connectors to interface payload data and communications with the Orbiter vehicle or payload specialist station displays.

c. Connectors to interface wideband data from the payload with a payload supplied recorder located in the Orbiter cabin; and,

d. Electrical power from the electrical power subsystem to the payload specialist station and the payloads.

3.3.1.3.3.5.5 Payload Data Processing. The Orbiter shall have the capability to checkout, monitor, and command payloads. The Orbiter must be capable of performing the checkout, monitor, and command functions at all times after lift-off. A capability for payload monitoring shall be provided for all flight phases and ground operations. Payload caution and warning signals shall be displayed to the flight crew and at the mission specialist station. The capability shall be provided to display payload parameters in realtime to the mission specialist station. To support this activity, the Orbiter shall provide the following computer capability: Essential functions will be provided for critical flight phases. For the on-orbit phase, a main memory capacity of 10,000 32-bit words, 18K equivalent computer adds per second shall be provided to perform these functions. The capability to overlay this 10,000 word segment of memory with programs from Orbiter mass storage shall be provided.

3.3.1.3.3.5.6 Recorder Interface. The Orbiter vehicle shall provide for the onboard recording of selected analog (such as IRIG frequency division multiplex) and digital payload scientific data.

3.3.1.3.3.6 Payload Thermal Control and Atmospheric Revitalization.

3.3.1.3.3.6.1 Thermal Control. The Orbiter vehicle shall provide a heat sink during all mission phases for the payload waste heat.

3.3.1.3.3.6.1.1 Payload Bay Doors Closed. During ascent (above approximately 100,000 ft altitude), entry (including Post Landing Thermal Conditions as specified in Paragraph 3.2.1.1.15) and on-orbit with payload bay doors closed, the heat removal capability from payload shall be 5200 BTU/hr with coolant temperatures of 45°F maximum to the payload and 100°F returned from the payload.

3.3.1.3.3.6.1.2 Payload Lay Doors Open. During orbital operations with the payload bay doors open, the ATCS will provide heat removal capability from the payload of 21,500 BTU/hr with coolant temperatures of 45°F maximum to the payload and 130°F returned from the payload.

3.3.1.3.3.6.1.3 Increased Heat Rejection Capability. During orbital operations with the payload bay doors open, the ATCS will provide a heat removal capability from payloads of 29,000 BTU/hr by the addition of a mission kit chargeable to the payload. Coolant temperatures will be 45°F maximum to the payload and 104°F from the payload if either Freon 21 or water is the payload fluid.

3.3.1.3.3.6.1.4 Interface Heat Exchanger. A single payload heat exchanger will be provided by the Orbiter to remove heat from the payloads. The heat exchanger will be sized to meet the 29,000 BTU/hr requirements of Paragraph 3.3.1.3.3.6.1.3.

3.3.1.3.3.6.1.5 Payload Coolant Fluid. The payload heat exchanger shall be designed so that any of the following can be selected (by the payloads) as a payload fluid: water, Freon 21.

3.3.1.3.3.6.1.6 Cabin Located Equipment. The Orbiter shall provide cooling for payload equipment located on the Orbiter aft flight deck. This cooling capacity shall be up to 0.75 kw average and 1.0 kw peak. Cooling requirements above 0.35 kw may be provided by reducing the concurrent heat removal capacity from payload equipment in the payload bay. The above values shall include up to 100 watts (341 BTU/hr) cooling for aft flight deck individual payload components consuming small quantities of power (10 watts) by direct radiation or convection to the cabin; specific forced air cooling shall not be required.

3.3.1.3.3.6.2 Atmospheric Revitalization. The Orbiter vehicle shall provide for atmospheric revitalization of habitable payload modules by providing accommodations for ducting for circulation of conditioned cabin air to the payload for use of up to four crewmen. The Orbiter ducting kit, which will provide this ducting, will be payload chargeable. Adequate air recirculation between the Orbiter cabin and the payload will be accomplished by a fan, sized and supplied as a part of and by the payload. The Orbiter shall also control and maintain module internal pressure. Expendables, related storage facilities and payload hardware required to accomplish these functions shall be chargeable to payload weight and volume. The Orbiter shall provide the capability for payloads to obtain oxygen from the Orbiter cryogenic oxygen tankage. To accomplish this, a regulated oxygen line shall be provided in the payload bay near the Orbiter aft cabin bulkhead.

3.3.1.3.3.7 Illumination. The Orbiter vehicle shall provide a lighting subsystem for illumination to support Orbiter/payload operations external to the Orbiter vehicle and inside the payload bay.

3.3.1.3.3.8 (Deleted).

3.3.1.3.3.9 Payload Conditioning Control. Capability for purging and atmospheric control of the payload bay, independent of the Orbiter vehicle internal structure, shall be provided by GSE while on the launch pad with the payload bay doors opened or closed. The Orbiter shall be designed for payload thermal conditioning and bay purging using conditioned purge gas. Air shall be used as the purge gas with the payload bay doors open and either air or GN_2 with the doors closed. Connectors and internal plumbing for payload conditioning and bay purging and atmospheric control, located outside of the payload bay clear volume, shall be chargeable to the Orbiter vehicle. The temperature control in excess of that provided by the Orbiter, vehicle atmospheric composition, and air filtration for a controlled payload environment on the launch pad shall be the responsibility of the GSE.

3.3.1.3.3.10 Payload Bay Acoustics. The Orbiter vehicle payload bay interior sound pressure level shall not exceed a maximum overall of 145 dB for the spectral frequency distribution shown in Figure 3.3.1.3.3.10 (TBD).

3.3.1.3.3.11 Pressure. The Orbiter vehicle payload bay shall be vented during launch and entry and shall operate unpressurized during the orbital phase of the mission. Venting and repressurization of the payload bay shall be separate from the rest of the vehicle vent/repressurization system.

3.3.1.3.3.12 Payload Bay Wall Temperatures. The internal wall temperatures for the payload bay are dependent on cargo element energy sources and sinks, cargo element configuration, and specific mission attitude timeline. Realistic payload bay environments require a detailed integrated Orbiter/cargo element thermal analysis.

3.3.1.3.3.13 Payload Bay Vibration. The Orbiter vehicle payload bay attachment vibration environments shall Not exceed those shown in Figure 3.3.1.3.3.13 (TBD) *

3.3.1.3.3.14 (Deleted).

3.3.1.3.3.15 (Deleted).

3.3.1.3.3.16 Attachments for Payload Deployment/Retrieval. The Orbiter manipulator(s) shall interface with the payload provided attach points for payload deployment and retrieval.

3.3.1.3.3.17 Orbiter/Payload Interface Connectors. The Orbiter vehicle shall provide standard connectors for electrical power, electrical signals, and fluid interfaces in the payload bay and aft flight deck.

3.3.1.3.3.18 Payload Heat Rejection Kit Mounting and Venting Interfaces. The Orbiter vehicle shall provide mounting capability and necessary vents for a payload supplied heat removal kit having the capacity to remove 48,000 Btu/hr.

3.3.1.3.3.19 Orbiter Interface with Spacelab. The Orbiter shall interface with the Spacelab as defined in the following ICDs.

3.3.1.3.3.19.1 Shuttle Vehicle/Spacelab Structural/Mechanical Interfaces. ICD 2-05101.

3.3.1.3.3.19.2 Shuttle Vehicle/Spacelab ECS/Thermal Interfaces. ICD 2-05201.

3.3.1.3.3.19.3 Shuttle Vehicle/Spacelab Avionics Interfaces. ICD 3-05301.

3.3.1.3.4 Orbiter Vehicle Interface with Main Engine. The Orbiter vehicle shall interface with the main engines as defined in SSME/Orbiter ICD 13M15000.

3.3.1.3.5 Orbiter Vehicle Interface with Carrier Aircraft. The Orbiter vehicle shall interface with the carrier aircraft as defined in the Orbiter/Carrier Aircraft Vehicle ICD 2-17001.

3.3.1.3.6 (Deleted).

3.3.1.3.7 Orbiter Vehicle Interface with Primary Landing Station. The Orbiter vehicle shall interface with the Landing Station as defined in Orbiter/Landing Station ICD 2-1A001.

3.3.1.3.8 Orbiter Vehicle Interface with Orbiter Processing Facility. The Orbiter shall interface with the Orbiter processing facility as defined in Orbiter/Processing Station ICD 2-1A002.

3.3.1.3.9 Orbiter Vehicle Interface with Hypergolic Maintenance and Checkout Station. The Orbiter shall interface with the Hypergolic Maintenance and Checkout Station as defined in Orbiter/Hypergolic Station ICD 2-1A003.

3.3.1.3.10 Orbiter Vehicle Interface with Shuttle Vehicle Assembly and Checkout Station. The Orbiter shall interface with the Shuttle Vehicle Assembly and Checkout Station in the VAB as defined in Shuttle System/VAB ICD 2-0A001.

3.3.1.3.11 Orbiter Vehicle Interface with Launch Pad Station. The Orbiter shall interface with the launch pad station as defined in Shuttle System/Launch Pad and MLP ICD 2-0A002.

3.3.1.3.12 Orbiter Vehicle Interface with Secondary Landing Stations. The Orbiter shall interface with the secondary landing station as defined in Orbiter/Landing Station (EAFB) ICD 2-1D003.

3.3.1.3.13 (Deleted).

3.3.1.3.14 Orbiter Vehicle Interface with Communications and Tracking Functions (DRFC). The Orbiter shall interface with the communications and tracking functions at DFRC as defined in Orbiter/Communications and Tracking (DFRC) ICD 2-1D001.

3.3.1.3.15 Orbiter and Carrier Aircraft Interface with the Mate Demate Device. The Orbiter and Carrier Aircraft shall interface with the mate demate device as defined in the Orbiter and Carrier Aircraft/Mate Demate ICD 2-1D004.

CARGO WEIGHT	DISTANCE FROM FORWARD END OF PAYLOAD BAY ENVELOPE Xo = 582.0 (INCHES)	
LBS. X 1000	FWD LIMIT	AFT LIMIT
3.0	-	698.76
3.5	-	625.13
5.0	-	595.68
6.5	-	579.82
8.0	32.44*	569.91
9.5	105.43	-563.12
11.0	158.51	558.19
12.5	198.35	554.44
14.0	230.55	551.50
15.5	256.11	549.12
17.0	277.16	547.17
18.5	294.80	545.53
20.0	309.79	544.13
21.5	322.69	542.94
23.0	333.91	541.89
24.5	343.75	540.98
26.0	352.46	540.17
27.5	360.22	539.45
29.0	367.18	538.80
30.5	373.45	538.22
32.0	379.13	537.69
33.5	384.31	537.21
35.0	389.04	536.77
36.5	393.38	536.37
38.0	397.38	536.00
39.5	401.07	535.65
41.0	404.50	535.33
42.5	407.68	535.04
44.0	410.65	534.76
45.5	413.42	534.51
47.0	416.01	534.26
48.5	418.45	534.04
50.0	420.73	533.83
51.5	422.89	533.63
53.0	424.92	533.44
54.5	426.84	533.26
56.0	428.66	533.09
57.5	430.38	532.93
59.0	432.02	532.78
60.5	433.57	532.63
62.0	435.05	532.50
63.5	436.46	532.36
65.0	437.80	532.24

*Fwd limit is zero at 7475.4 lb.

Table 3.3.1.2.1.2.2.a Cargo Y Center-of-Gravity Envelope

CARGO WEIGHT LBS. X 1000	DISTANCE FROM PAYLOAD BAY C Yo = 0 (INCHES)
2	±32.82
4	±17.16
6	±11.94
8	± 9.33
10	± 7.76
12	± 6.72
14	± 5.97
16	± 5.42
18	± 4.98
20	± 4.63
22	± 4.35
24	± 4.11
26	± 3.91
28	± 3.74
30	± 3.59
32	± 3.46
34	± 3.34
36	± 3.24
38	± 3.15
40	± 3.07
42	± 2.99
44	± 2.92
46	± 2.86
48	± 2.81
50	± 2.75
52	± 2.70
54	± 2.66
56	± 2.62
58	± 2.58
60	± 2.54
62	± 2.51
64	± 2.48
65	± 2.46

Table 3.3.1.2.1.2.2b Cargo Z Center-of-Gravity Envelope

CARGO WEIGHT LBS. X 1000	DISTANCE FROM PAYLOAD BAY CENTERLINE Zo = 400 (INCHES)			
	CARGO UPPER LIMIT	UPPER LIMIT FOR PAYLOAD MOUNTED ON PAYLOAD BAY ATTACHMENTS	LOWER LIMIT FOR PAYLOAD MOUNTED ON PAYLOAD BAY ATTACHMENTS	CARGO LOWER CARGO
2	90.00	90.00	-90.00	-90.00
3	90.00	90.00	-90.00	-90.00
4	90.00	90.00	-45.00	-90.00
5	90.00	45.00	-45.00	-90.00
6	90.00	45.00	-45.00	-90.00
8	90.00	45.00	-45.00	-90.00
10	90.00	45.00	-45.00	-90.00
12	90.00	45.00	-45.00	-90.00
14	79.59(1)	45.00	-45.00	-90.00
16	67.70	45.00	-45.00	-90.00
18	58.46	45.00	-45.00	-90.00
20	51.06	45.00	-45.00	-90.00
22	45.01	45.00	-45.00	-90.00
24	39.97	39.97(3)	-45.00	-90.00
26	35.70	35.70	-45.00	-90.00
28	32.04	32.04	-45.00	-90.00
30	28.87	28.87	-45.00	-90.--
32	26.10	26.10	-45.00	-88.94(2)
34	23.65	23..65	-43.48	-86.06
36	21.48	21.48	-41.97	-83.50
38	19.53	19.53	-40.45	-81.22
40	17.78	17.78	-38.94	-77.29
42	16.20	16.20	-37.42	-77.29
44	14.76	14.58(4)	-35.91	-75.59
46	13.44	12.48	-34.39	-74.05
48	12.23	10.59	-32.88	-72.63
50	11.12	8.91	-31.36	-71.32
52	10.10	7.44	-29.85	-70.12
54	9.15	6.18	-28.33	-69.00
56	8.27	5.13	-26.82	-67.97
58	7.45	4.29	-25.30	-67.00
60	6.69	3.66	-23.79	-66.10
62	5.97	3.24	-22.27	-65.26
64	5.30	3.03	-20.76	-64.47
65	4.98	3.00	-20.00	-64.09

(1)Cargo CG reaches upper cargo envelope limit of 90 in. at 12,618.4 lb.
(2)Cargo CG reaches lower cargo envelope limit of -90 in. at 31,323.4 lb.
(3)Cargo CG intersects upper control limit line at 22004.0 lb.
(4)Cargo CG intersects upper control limit line at 43571.8 lb.

Table 3.3.1.3.3.12 (Deleted)

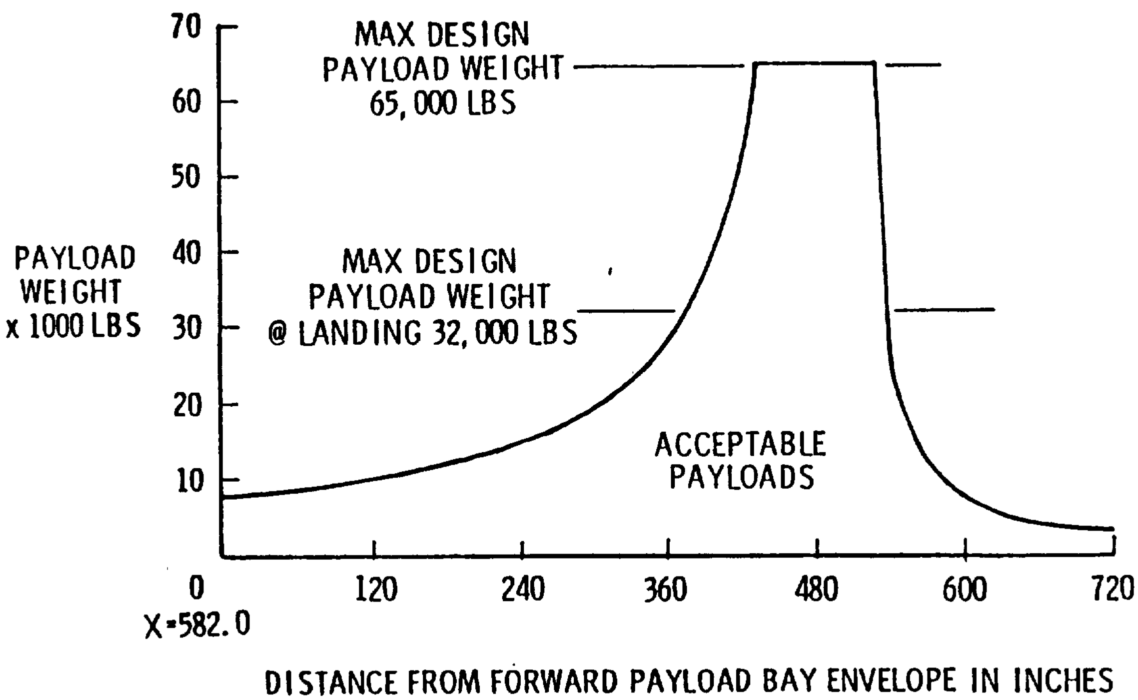


Figure 3.3.1.2.1.2.2 Payload CG Limits, (Along X-Axis)

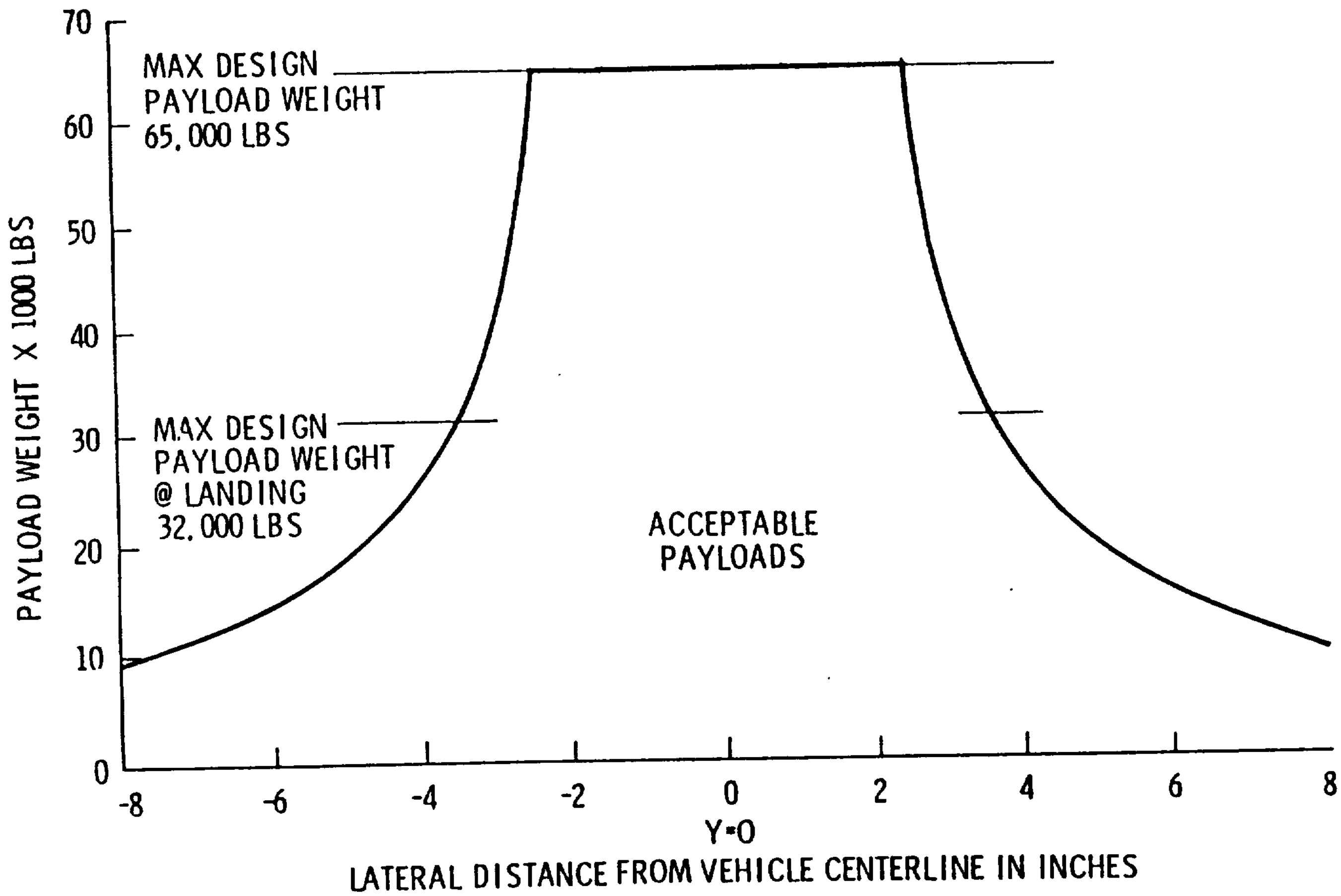


Figure 3.3.1.2.1.2.2a Payload CG Limits, (Along Y-Axis)

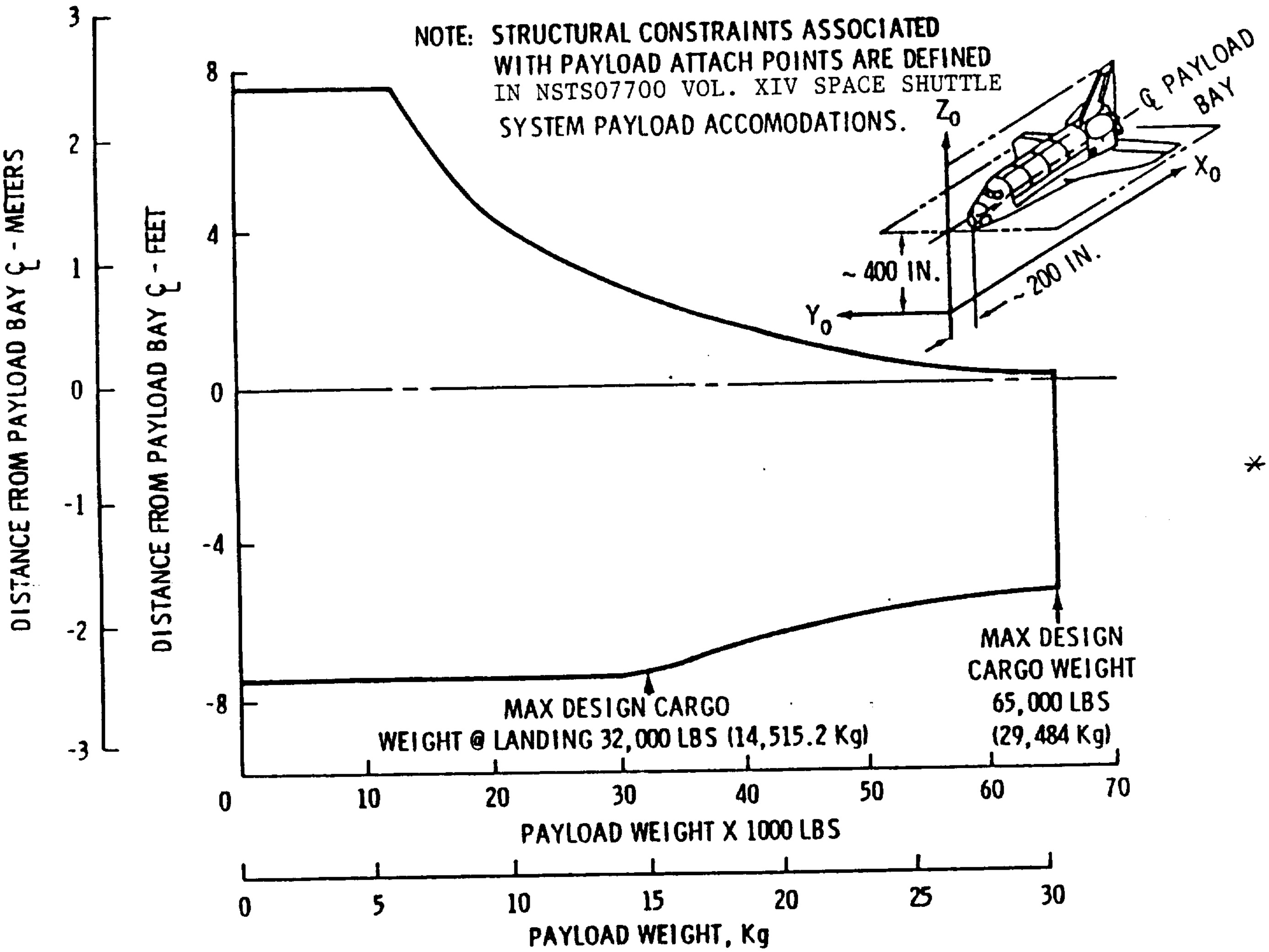


Figure 3.3.1.2.1.2.2b Payload CG Limits, (Along Z-Axis)

Figure 3.3.1.3.3.10 Spectral Frequency Distribution

(TO BE DETERMINED)

Figure 3.3.1.3.3.13 Payload Bay Attachment Vibration Environments

(TO BE DETERMINED)

3.3.2 Liquid Rocket Booster (LRB) Characteristics. *

3.3.2.1 LRB Performance Characteristics. *

3.3.2.1.1 LRB Ascent. The LRBs when operating in a normal mode in parallel with the Orbiter vehicle MEs, shall provide impulse and thrust vector control to thrust the flight vehicle from lift-off to LRB staging from the ETR and WTR launch sites. *

3.3.2.1.2 Liquid Rocket Booster (LRB) Performance Requirements *
The LRB shall be designed to provide the following Shuttle performance when launched from ETR to 28.5° inclination, 110 NM orbit with SSME power level limited to 104% (109% for abort);

Nominal design case:	75,500 lb payload
Alternate design case:	67,500 lb payload

3.3.2.1.3 LRB Temperature Limits. *

3.3.2.1.4 Performance Sizing. (TBD) *

3.3.2.1.5 Design. (TBD) *

3.3.2.1.6 Operational Performance Capability. (TBD) *

3.3.2.1.7 Center-of-Gravity. (TBD) *

3.3.2.2 LRB Design Characteristics. *

3.3.2.2.1 External Configuration. The LRB shall conform to the moldline envelope specified in (TBD) , "Shuttle Moldlines and Protuberances." *

3.3.2.2.1.1 LRB Reuse. (TBD) *

3.3.2.2.1.2 LRB Control Weight. The LRB inert control weight is specified in (TBD) *

3.3.2.2.1.3 LRB Propellant Control Weight. The control weights for usable and residual propellants are specified in (TBD) *

3.3.2.2.1.4 Thrust Vector Control (TVC). (TBD) *

a. Gimbal Axis Orientation - The LRB TVC gimbal axis shall be oriented at (TBD) degrees about and perpendicular to the longitudinal LRB axis. *

b. Gimbal Angle - The LRB TVC subsystem shall be capable of providing the gimbal angles shown below:

	LRB Nozzle Stagnation Pressure (psi)	Nominal Gimbal Angle (degrees)
Actuator		
Actuator Extend Direction		
	(TBD)	
Actuator Retract Direction		

*

Three sigma variation shall be less than ±10 percent of the gimbal angle. Dual actuator vector angles are

$\sqrt{\theta_1{}^2 + \theta_2{}^2}$ where θ_1 and θ_2 are nominal gimbal angles obtained from table above. Linear interpolation shall be used to obtain values for intermediate pressures. Effects of nozzle pivot shift, flexibility of the bearing and structures, null shift, and alignment error shall be included in the above requirement. *

Nozzle null offset angle(s) - (TBD) *

Commands to the actuators will be limited by Orbiter software to avoid commanding more than (TBD) degrees gimbal angle in the plane of the actuator for LRB pressures below 50 psi. *

c. Gimbal Rate - (TBD) *

d. Angular Acceleration - Net angular acceleration capability of the LRB TVC with rated disturbance loads in opposition shall be at least (TBD) radians/sec^2. *

e. Phase Lag - (TBD) *

f. Step Response - (TBD) *

g. Command Channel Bypass - (TBD) *

h. Fault Detection - The TVC actuators shall provide interfacing instrumentation compatible with the fault detection isolation and recovery (FDIR) electronics. *

3.3.2.2.2 LRB Ignition System. The LRB ignition system shall have the capability to be remotely safed or armed from the launch control center. *

3.3.2.2.3 LRB Destruct System. The LRBs shall be provided with ground-commanded systems to destruct the LRBs. System components shall be reusable where cost savings will result. *

3.3.2.3 LRB Interface Characteristics. *

3.3.2.3.1 LRB/Orbiter Interface. Functional interfaces for control and instrumentation between the LRB and the Orbiter vehicle are covered in (TBD) *

3.3.2.3.2 LRB/ET Interface. The LRB shall interface with the ET as defined in ICD 2-24001. *

3.3.2.3.2.1 LRB/ET Umbilical. An umbilical shall provide a conductive signal path between the LRB and the ET for the following electrical signals:

a. LRB Status Monitor Signals

b. Sequence Commands

c. (TBD) *

3.3.2.3.3 LRB Interface with Shuttle Vehicle Assembly and Checkout Station. The LRB shall interface with the Shuttle Vehicle Assembly and Checkout Station as defined in (TBD) *

3.3.2.3.4 LRB Interface with LRB Processing and Storage Station. The LRB shall interface with the LRB processing and storage station as defined in (TBD) *

3.3.2.2.5 LRB Interface with LRB Retrieval and Disassembly Station. The LRB shall interface with the LRB retrieval and disassembly station as defined in (TBD) *

3.3.2.3.6 LRB Interface with Launch Pad Station. The LRB shall interface with the launch pad station as defined in (TBD) *

3.3.2.3.7 LRB Interface with LRB/Refurbishment and Subassembly Station. The LRB shall interface with the LRB Refurbishment and Subassembly Station as defined in the (TBD) *

3.3.2.4 Ullage Volume (TBD) *

3.3.2.5 Structural Stability (TBD) *

3.3.2.6 Preparation and Servicing (TBD) *

3.3.2.7 Propellant Management Instrumentation (TBD) *

3.3.2.8 Propellant Loading (TBD) *

3.3.2.9 Propellant Depletion Sensors (TBD) *

3.3.2.10 Ullage Pressure (TBD) *

3.3.2.11 Propellant Slosh Damping (TBD) *

3.3.2.12 Handling (TBD) *

3.3.2.13 Thermal Protection (TBD) *

3.3.2.14 Propellant Dispersal System (TBD) *

3.3.2.15 LRB Engine Characteristics *

Table 3.3.2.1.2a Standard LRB Nominal Thrust-Time Limits
(Vacuum - $60^{o}F$)

TO BE DETERMINED

*

Table 3.3.2.1.2a Standard LRB Nominal Thrust-Time Limits (Vacuum - 60°F) - Continued

TO BE DETERMINED

*

Table 3.3.2.1.2a Standard LRB Nominal Thrust-Time Limits (Vacuum - 60°F) - Concluded

TO BE DETERMINED

*

Table 3.3.2.1.2b Alternate LRB Nominal Thrust-Time Limits
(Vacuum - 60°F)

TO BE DETERMINED

*

Table 3.3.2.1.2b Alternate LRB Nominal Thrust-Time Limits (Vacuum - $60^{o}F$) - Continued

TO BE DETERMINED

*

Table 3.3.2.1.2b Alternate LRB Nominal Thrust-Time Limits (Vacuum - $60^{o}F$) - Concluded

TO BE DETERMINED

*

C2

(TBD) *

Figure 3.3.2.1.2a Standard LRB Nominal Performance Requirements (Vacuum $60^{\circ}F$)

(TBD) *

Figure 3.3.2.1.2b Alternate LRB Nominal Performance Requirements
(Vacuum $60^{o}F$)

(TBD) *

Figure 3.3.2.1.2c LRB System Thrust Vector Alignment

(TBD) *

Figure 3.3.2.1.2d Thrust Imbalance

(TBD) *

Figure 3.3.2.1.2e Ignition Thrust Imbalance

(TBD) *

Figure 3.3.2.1.2f Steady State Thrust Imbalance

3.3.3 External Tank (ET) Characteristics.

3.3.3.1 External Tank Performance Characteristics.

3.3.3.1.1 External Tank Mass Data.

3.3.3.1.1.1 ET Size. The ET shall conform with the moldline envelope specified in ICD 2-00001, "Shuttle Moldlines Protuberances," and be sized to accommodate the main stage propellant loading specified in NSTS 07700, Volume X, Appendix 10.12. *

3.3.3.1.1.2 ET Control Weight. The External Tank inert control weight is specified in Appendix 10.12. The control weights for usable propellants and gaseous residuals in the ET and associated lines are specified in NSTS 07700, Volume X, Appendix 10.12. *

3.3.3.1.1.3 Center-of-Gravity. The Heavyweight External Tank center-of-gravity shall be as specified below:

CONDITION	LONGITUDINAL (X_{ET})	LATERAL (Y_{ET})	VERTICAL (Z_{ET})
Inert	1370.0 in ± 7.5	2.2 in ± 5.0	427.2 in ± 5.0

The Lightweight External Tank (6000 lbs. weight reduction) center of gravity shall be as specified below:

CONDITION	LONGITUDINAL (X_{ET})	LATERAL (Y_{ET})	VERTICAL (Z_{ET})
Inert	1347.0 in ± 7.5	2.5 in ± 5.0	425.2 in ± 5.0

3.3.3.1.1.4 ET Ullage Volume. The ET tank ullage volume shall be a minimum of 2.65% and 2.0% of the LO_2 and LH_2 volumes respectively, load at engine start command.

3.3.3.2 External Tank Design Characteristics.

3.3.3.2.1 Structural Stability. The ET Structure shall not require pressurization for stability or GSE support for the attached Orbiter during ground handling, transportation, or while on the launch pad in either a fueled or unfueled condition with the exception of the propellant tanks, which may be designed to require pressure stabilization during fill and drain operations, but shall not require pressure stabilization during replenish operations.

3.3.3.2.1.1 (Deleted).

3.3.3.2.1.2 (Deleted).

3.3.3.2.2 Preparation and Servicing. The external tank preparation and servicing, excluding final servicing, shall be completed prior to standby status (see 6.1.1).

3.3.3.2.3 Propellant Management Instrumentation. Measurements shall be provided to accommodate propellant loading, mainstage tank pressurization and LH_2 depletion, to satisfy the requirements of Paragraphs 3.2.1.2.8, 3.2.2.1.13 and ET pressurant flow requirements.

3.3.3.2.3.1 Propellant Loading. The vehicle shall provide level indications for propellant loading visibility of the LH_2 and LO_2 tanks.

3.3.3.2.3.2 LH_2 Propellant Depletion Sensors. The LH_2 tank shall provide propellant depletion signals to the Orbiter for Orbiter SSME cutoff.

3.3.3.2.3.3 Ullage Pressure. The ET shall provide signals of LO_2 and LH_2 ullage pressure to the Orbiter. The ET shall provide two low range (0-5 psig) LO_2 ullage pressure measurements which utilize ground power and ground readout for the LPS.

3.3.3.2.4 Propellant Slosh Damping. The external tank shall provide slosh damping in the LO_2 tank. The mission conditions to be considered in establishing propellant loading are nominal conditions for the design reference missions specified in Paragraph 3.2.1.1.3. The LOX slosh damping ratio for the lightweight ET with four slosh baffles in the LOX tank shall meet the following requirements:

a. Greater than .01 for a fluid level between the bottom of the tank and 230 inches.

b. Greater than .005 for a fluid level between 230 inches and 300 inches.

c. Greater than or equal to .002 for a fluid level greater than 300 inches.

Figure 3.3.3.2.4 defines the minimum slosh damping required for the lightweight tank and reflects the prediction of the slosh damping to meet or exceed this requirement.

3.3.3.2.5 Handling. The External Tank with insulation shall be capable of being hoisted, erected, transported, handled, etc., and cryogenically loaded and unloaded after verification activities are complete without requiring External Tank insulation inspection or special verification.

3.3.3.2.6 Thermal Protection. The External Tank shall incorporate thermal protection, as required, to satisfy all functional and performance requirements within the design environments specified in Paragraph 3.2.2.1.17 and minimize the formation of ice as specified in NSTS 16007.

Note: Until the atmospheric requirements are established, a minimum of one inch of SOFI shall be added to the external surface of the tank to satisfy this requirement.

3.3.3.2.7 ET Propellant Dispersal System. The ET shall be provided with a ground-commanded system to disperse the ET propellants. The components shall be installed so as to be readily added or removed, where possible. The system will be protected against any auto-detonation to 255,000 feet for the mission 3A-AOA ET entry.

3.3.3.2.8 Safe Separation Distance. To insure a minimum safe separation distance between the Orbiter and ET of 4 N.MI., the ET shall not rupture during the first 225 seconds following MECO for the most severe ascent heating design mission (Mission 3A-AOA). For nominal mission (i.e., non-abort) the ET shall not rupture until after entry below 350,000 feet.

3.3.3.2.9 (Deleted).

3.3.3.3 External Tank interface Characteristics.

3.3.3.3.1 External Tank Interface with Orbiter. See Paragraph 3.3.1.3.1 for interface requirements.

3.3.3.3.2 External Tank Interface with Liquid Rocket Booster. See Paragraph 3.3.2.3.2 for interface requirements. *

3.3.3.3.3 External Tank Interface with Shuttle Vehicle Assembly and Checkout Station. The External Tank shall interface with the Shuttle Vehicle Assembly and Checkout Station as defined in Shuttle System/VAB ICD 2-0A001.

3.3.3.3.4 External Tank Interface with ET Processing and Storage Station. The External Tank shall interface with the ET Processing and Storage Station as defined in External Tank/Receiving, Storage and Checkout Station ICD 2-2A001.

3.3.3.3.5 External Tank Interface with Launch Pad Station. The External Tank shall interface with the Launch Pad Station as defined in Shuttle Systems/Launch Pad and MLP ICD 2-0A002.

Table 3.3.3.2.4 (Deleted)

Figure 3.3.3.1.1.1 (Deleted)

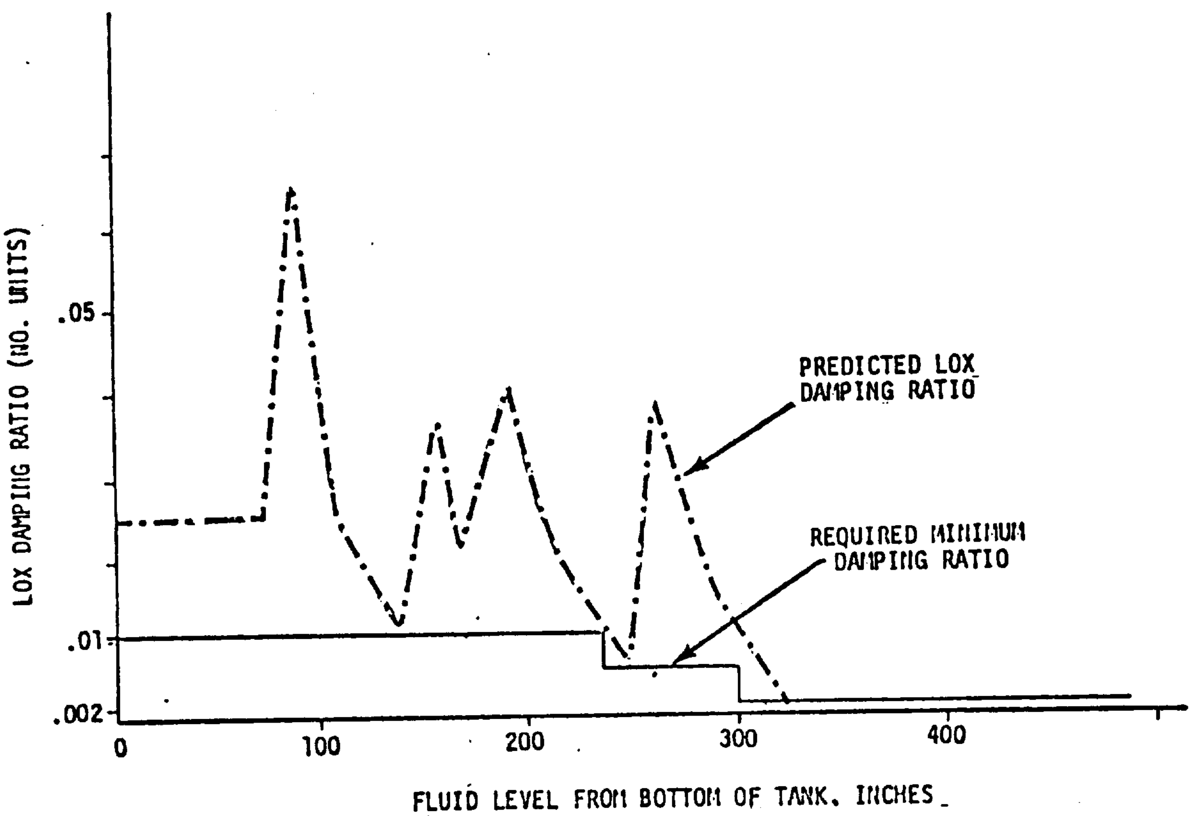


Figure 3.3.3.2.4 Minimum Slosh Damping Requirements for Lightweight tank

THIS PAGE INTENTIONALLY LEFT BLANK

3.3.4 Main Engine Characteristics. The Space Shuttle Main Engine (SSME) shall meet the requirements specifed below and in ICD No. 13M15000.

3.3.4.1 Main Engine Performance Characteristics.

3.3.4.1.1 Engine Performance Levels. The following single engine performance shall be provided by the SSME.

Parameters	Vacuum	Sea Level
Thrust (Klb)*	470 ± 6	375 ± 6
Mixture Ratio*	6.0 ± 1%	6.0 ± 1%
Specific Impulse*	455.2 ± 2.3	363.2±1.8
Minimum Engine Guaranteed Isp	452.9	361.4

* Based on 3 sigma precision

3.3.4.1.2 Engine Operating Conditions. The engine operating conditions for thrust, specific impulse, and mixture ratio are given in Table 3.3.4.1.2.

3.3.4.2 Main Engine Design Characteristics.

3.3.4.2.1 Main Engine Control Weights. The main propulsion engine control weight is specified in NSTS 07700, Volume X, Appendix 10.12. *

3.3.4.2.2 Main Engine Hydraulic Design. Hydraulic subsystem design and installation, shall be in accordance with MIL-H-5440. This specification (MIL-H-544) shall take precedence over safety factors stated in Paragraph 3.2.2.1.5.2.

3.3.4.2.3 Main Engine Flight Acceleration Safety Cutoff System. The SSME shall provide a capability to monitor high pressure turbopump vibration and initiate a safe engine shutdown in the event safety critical vibration conditions are encountered. The system design shall not be less than two fault tolerant (assures shutdown capability after one failure and precludes inadvertent shutdown with two failures) when operated in an active shutdown mode.

3.3.4.3 Main Engine Interface Characteristics.

3.3.4.3.1 Main Engine Interface with Orbiter. See Paragraph 3.3.1.3.4 for interface requirements.

Table 3.3.4.1.2 Main Engine Operating Conditions

	Sea Level	Vacuum
Full Power Level (FPL) (1/2)		
Thrust, lbf	417,300	512,300
Specific Impulse, (lbf-sec)/lbm (4)	_ 369.0	_ 453.0
Mixture Ratio* (5)	6.0±1%	6.0±1%
Rated Power Level (RPL) (1/2)		
Thrust, lbf	375,000	470,000
Specific Impulse, (lbf-sec)/lbm (4)	_ 361.4	_ 452.9
Mixture Ratio* (5)	6.0±1%	6.0±1%
Minimum Power Level (MPL) (1/2)		
Thrust, lbf	See Note (3)	235,000
Specific Impulse, (lbf-sec)/lbm (4)		_ 451.1
Mixture Ratio* (5)		6.0 ± 1%

* Based on 3 Sigma Precision

Notes:

(1) Power level is variable in 1% increments (47000 lbs) of RPL from FPL to MPL by electrical signal.

(2) All values referenced to the propellant inlet condition ranges specified in Figure 4.2..1-1 of Orbiter Vehicle/Space Shuttle Main Engine, ICD 13M15000.

(3) The engine will be capable of being operated at sea level conditions without the use of altitude test facilities or restrainer arms at RPL and above.

(4) All values referenced to the nominal pressurant tapoffs.

(5) The mixture ratio shall be set to assessment values of 6.026 (ET-11 thru ET-16), 6.0117 (ET-17 thru ET-22), 6.0227 (ET-23 thru ET-39), and 6.0317 (ET-40 and subs) with an uncertainty of ± 1%. These adjustments shall not be construed as changes to SSME design, performance or certification requirements.

(6) Off nominal SSME performance data will be supplied for application to realtime inflight failure detection and ascent performance analysis. These data will include changes in thrust, mixture ratio, and engine Specific Impulse (Isp). The data will be of sufficient accuracy to permit use by the Abort Region Determinator (ARD) for abort decisions to insure flight safety.

3.3.5 Carrier Aircraft Characteristics

3.3.5.1 Carrier Aircraft Performance Characteristics.

3.3.5.1.1 Carrier Aircraft Ferry Range. The carrier aircraft shall be capable of ferrying the Orbiter vehicle non-stop from Edwards AFB to Kennedy Space Center.

3.3.5.1.2 Carrier Aircraft/Orbiter Flight Test. The carrier aircraft shall be modified so as to be capable to perform as a platform for air launch of the Orbiter during Approach and Landing Tests (ALT). The carrier aircraft shall have the capability for an aborted air launch and return to landing with the Orbiter in the air launch attitude.

3.3.5.2 Carrier Aircraft Design Characteristics.

3.3.5.2.1 Structure.

3.3.5.2.1.1 Ferry Configuration. The carrier aircraft shall be modified to provide ferry capability of the Orbiter vehicle. On a selected basis, subject to Level II approval, payloads may be ferried to the launch site in the Orbiter payload bay.

3.3.5.3 Carrier Aircraft Interface Characteristics.

3.3.5.3.1 Carrier Aircraft/Orbiter. See Paragraph 3.3.1.3.5 for interface requirements.

3.3.5.3.2 Carrier Aircraft Interface with Dryden Flight Research Center, Spaceflight Tracking and Data Network for ALT. Radio Frequency Communications and Tracking functions between the Shuttle Carrier Aircraft and Dryden Flight Research Center, Spaceflight Tracking and Data Network, and chase planes are controlled by ICD 2-7D001, "JSC/GSFC/DFRC SCA RF Communications and Tracking".

3.3.6 Mission Equipment Kit Characteristics

3.3.6.1 Mission Equipment Kit Performance Characteristics. A mission kit is flight hardware that either extends the capability of the Space Shuttle Vehicle or provides an interface to the cargo. Performance characteristics of individual mission equipment kits will depend on the kit's function and will be defined by the procuring element project office to the supplying contractor and shall be incorporated in the Contractor's End Item (CEI) specifications.

3.3.6.2 Mission Equipment Kit Design Criteria. All mission equipment kits shall be designed to conform to the following general design criteria.

a. Installation/removal, reconfiguration or maintenance time required for turnaround between missions shall be minimized.

b. The capability for installation/removal, reconfiguration or maintenance in both the norizontal (preparation facility) or vertical (launch pad) shall be considered.

c. The parts list breakdown (details, subassembly or assembly) shall be structured to consider usage of all or portions of the mission kits to provide multiple mission options and to allow the option to fly unneeded portions of the kits as "scar" weight.

d. Kit designs which lend themselves to partial usage shall contain the loose hardware (caps, plugs, covers, etc.) for achieving the partial flight configuration.

3.4 GROUND OPERATIONS COMPLEX PERFORMANCE AND DESIGN CHARACTERISTICS.

3.4.1 Primary Landing Station (KSC).

3.4.1.1 Performance Characteristics. Capabilities shall be provided for:

a. Support of Orbiter vehicle operations starting at post black-out through approach and rollout under all weather conditions

b. Orbiter vehicle and conventional aircraft flight crew and ground crew egress and ingress

c. Orbiter vehicle post-landing operations, including deactivation, securing, safing, cooling, contingency DC power, and air purging

d. Communications between the Orbiter and a ground control/operations center

e. Communications between the Orbiter/conventional aircraft and ground crews

f. Conventional aircraft, landing, and takeoff

g. Servicing and takeoff/landing operations for the Orbiter and carrier aircraft configuration

h. Payload RTG cooling

3.4.1.2 Design Characteristics.

3.4.1.2.1 Runway Size. The runway for KSC shall be 15,000 feet long with 1,000 foot overruns, 300 feet wide plus 50 foot stabilized shoulders. The stabilized shoulders shall not contribute debris to the runway.

3.4.1.2.2 Landing Runway Loads. For purposes of runway design, the following criteria shall be used:

Design Wheel Load - Static	60,000 lbs.
Design Landing Weights	
Nominal Operations	215,000
Abort Operations	265,000
Tire Pressure	230-385 psi

Main Gear Loads	90% Gross Vehicle Weight
Main Gear Spacing	23 ft. Center-to-Center
Main Gear Tire Spacing	36 inch Center-to-Center
Orbiter Runway Operations	600

3.4.1.2.3 Runway Surface. The runway surface shall be surface treated to minimize hydroplaning.

3.4.1.2.4 Taxiway. Taxiway capable of accommodating the Orbiter shall be provided between the landing area and the Orbiter maintenance and checkout facility station.

3.4.1.2.5 Landing Aids. The primary landing aids to be installed will be a completely redundant microwave landing system at each end of runway (RW 15 & 33) and TACAN located close to the runway.

Visual Landing aids will be provided for day and night operations to include the following:

a. Provisions for flood lights to illuminate touchdown and roll out area

b. Approach and centerline lighting or a transition light to allow final approach guidance and roll out control

c. Outer glide slope aim point at the intercept point of the ground and outer glide slope

d. Precision Approach Path Indicator (PAPI) lights at the OGS intercept point

e. Ball/Bar inner glide slope at the intersection of the touchdown zone and 1 1/2 degree glide slope

f. Provision will be made to reduce power for all lighted aids to accommodate night operations

g. The location of these aids will be as identified in the JSC 16895 (Space Transportation System Navigation Aids List)

3.4.1.2.6 Runway and High Intensity Approach Lighting Systems. A high intensity approach lighting system at each end of the runway, and runway lighting system are required to support Orbiter instrument and night landings in limited weather conditions.

3.4.1.2.7 Instrumentation Systems. MSBLS-GS commissioning and flight inspections and determination of Orbiter touchdown point will be required.

3.4.1.2.8 Meteorological Measurement Equipment. Meteorological measurement equipment will be required.

3.4.1.2.9 (Deleted).

3.4.1.2.10 S-Band Ranging. Ranging information shall be initiated by the ground station, turned around by the Orbiter S-Band communication equipment, and transmitted back to the ground station.

3.4.1.2.11 Orbiter/Ground Communications. An S-Band and a UGF voice communications system will be required to handle operational communications between the ground and the Orbiter through rollout.

3.4.1.2.12 Orbiter/Ground Telemetry and Command. S-Band telemetry from the Orbiter to the ground and S-Band command capability from the ground to the Orbiter will be required.

3.4.1.2.13 (Deleted).

3.4.1.2.14 Landing and Post-landing Television. Live, color television shall be provided of landing and post-landing activities. Individual camera feeds shall be distributed to news media locally at KSC. A switched, best-source picture shall be transmitted to all participating NASA Centers.

3.4.1.3 Interface Characteristics. The Primary Landing Station shall interface with the Orbiter as defined in the Orbiter/Landing Station ICD 2-1A001.

3.4.2 <u>Secondary and Alternate Landing Stations</u>/

3.4.2.1 Secondary Landing Station (Edwards AFB).

3.4.2.1.1 Performance Characteristics. Capabilities shall be provided for:

a. Support of Orbiter vehicle operations starting at post black-out through approach and landing under all weather conditions

b. Orbiter vehicle and conventional aircraft flight crew and ground crew egress and ingress

c. Orbiter vehicle post-landing operations, including deactivation, deservicing, securing, safing, cooling, and towing

d. Communications between the Orbiter and a ground control/operations center

e. Communications between the Orbiter/conventional aircraft and ground crews

f. Servicing and takeoff/landing operations for the Orbiter and carrier aircraft configuration

g. Payload safing and/or removal and

h. Visual observations of the landing site for exercising rescue, safety, and support of Orbiter and conventional aircraft operations

On a selected basis, subject to Level II approval, payloads may be ferried to the launch site in the Orbiter payload bay.

3.4.2.1.2 Design Characteristics.

3.4.2.1.2.1 Runway. The runway shall be as presently exists at the secondary landing site.

3.4.2.1.2.2 Taxiways. Taxiways capable of accommodating the Orbiter shall be provided between the landing area and the Orbiter maintenance and safing area.

3.4.2.1.2.3 Landing Aids. The primary landing aids to be installed will be microwave landing system at each end of the runway. In addition, TACAN will be required.

Visual Landing aids will be provided for day and night operations to include the following:

a. Provision for flood lights to illuminate touchdown and roll out area

b. Approach and centerline lighting or a transition light to allow final approach guidance and roll out control

c. Outer glide slope aim point at the intercept point of the ground and outer glide slope

d. Precision Approach Path Indicator (PAPI) lights at the OGS intercept point

e. Ball/Bar inner glide slope at the intersection of the touchdown zone and 1 1/2 degree glide slope

f. Provision will be made to reduce power for all lighted aids to accommodate night operations

g. The location of these aids will be as identified in the JSC 16895 (Space Transportation System Navigation Aids List)

3.4.2.1.2.4 Runway Lighting Systems. The lighting system shall be as presently exists at the secondary landing site.

3.4.2.1.2.5 Instrumentation Systems. Metric tracking and airborne landing system calibration will be required.

3.4.2.1.2.6 Meteorological Measurement Equipment. Meteorological measurement equipment will be required.

3.4.2.1.2.7 (Deleted).

3.4.2.1.2.8 S-Band Ranging. Ranging information shall be initiated by the ground station, turned around by the Orbiter S-Band communication equipment, and transmitted back to the ground station.

3.4.2.1.2.9 Orbiter/Ground Communications. An S-Band and a UGF voice communications system will be required to handle operational communications between the ground and the Orbiter through rollout.

3.4.2.1.2.10 Orbiter/Ground Telemetry and Command. S-Band telemetry from the Orbiter to the ground and S-Band command capability from the ground to the Orbiter will be required.

3.4.2.1.2.11 Landing and Post-Landing Television. Live, color television shall be provided of landing and post-landing activities. Individual camera feeds shall be distributed to news media locally at Ames-DFRF. A switched, best-source picture shall be transmitted to all participating NASA Centers.

3.4.2.1.3 Interface Characteristics. The Seconday Landing Station shall interface with the Orbiter as defined in Orbiter/Landing Station (EAFB) ICD 3-1D003.

3.4.2.2 Alternate Landing Stations.

3.4.2.2.1 Alternate Landing Station White Sands Space Harbor (WSSH).

3.4.2.2.2 Performance Characteristics. In the event Edwards AFB is unavailable for an Orbiter landing, the following capabilities shall be provided: (1) within three hours of notification (a) support of Orbiter vehicle operations starting at post black-out through approach and landing for daylight operations (b) communications between the Orbiter and a ground control/operations center, (c) visual observation of the landing site for exercising rescue, safety, and support of the Orbiter; (2) within two weeks of notification (a) all capability defined in (1) above, (b) Orbiter vehicle flight crew egress and ground crew ingress, (c) Orbiter vehicle ground power, cooling purge and PRSD venting; (3) within 5 weeks of notification (a) all capability defined in (1) & (2) above, (b) Orbiter vehicle post landing deactivation, deservicing, securing, safing and towing, (c) Orbiter preparatin for ferry, (d) loading of Orbiter on Shuttle Carrier Aircraft, (e) servicing and take/off landing operations for Shuttle Carrier Aircraft. On a selected basis, subject to Level II approval, payloads may be ferried to the launch site in the Orbiter payload bay.

3.4.2.2.3 Design Characteristics.

3.4.2.2.4 Runway. A runway suitable for the Orbiter and at least 20,000 feet long and 300 feet wide shall be provided. The runway shall be aligned with the 35-17 compass bearing.

3.4.2.2.5 Taxiway. Taxiways capable of accommodating the Orbiter shall be provided between the landing area and the Orbiter maintenance and safing area.

3.4.2.2.6 Landing Areas. The primary landing aids to be installed will be completely redundant microwave landing system on RW17 (Northrup Strip) and a mobile TACAN located in close proximity to the runway.

Visual Landing aids will be provided for day and night operations to include the following:

a. Provision for flood lights to illuminate touchdown and roll out area

b. Approach and centerline lighting or a transition light to allow final approach guidance and roll out control

c. Outer glide slope aim point at the intercept point of the ground and outer glide slope

d. Precision Approach Path Indicator (PAPI) lights at the OGS intercept point

e. Ball/Bar inner glide slope at the intersection of the touchdown zone and 1 1/2 degree glide slope

f. Provision will be made to reduce power for all lighted aids to accommodate night operations

g. The location of these aids will be as identified in the JSC 16895 (Space Transportation System Navigation Aids List)

3.4.2.2.7 Meteorological Measurement Equipment. Meteorological measurement equipment will be required.

3.4.2.2.8 (Deleted).

3.4.2.2.9 Orbiter/Ground Communication. A VHF voice communications system will be required to handle operational communications between the ground and the Orbiter through rollout.

3.4.2.2.10 (Deleted).

3.4.2.3 Trans Atlantic Landing Station (TAL).

3.4.2.3.1 Trans Atlantic Landing Station (Dakar, Senegal and Rota, Spain).

3.4.2.3.2 Performance Characteristics. Capabilities shall be provided for:

a. Support of Orbiter vehicle operations starting at post black-out through approach and landing under all weather conditions.

b. Rescue, safety and medical operations

c. Crew egress

d. Communications between Orbiter and ground operations

e. Post-landing towing and securing

On a selected basis, subject to Level II approval, payloads may be ferried to the launch site in the Orbiter payload bay.

3.4.2.3.3 Design Characterization.

3.4.2.3.4 Runway. The runway shall be as presently exists at the TAL site.

3.4.2.3.5 Landing Aids. The primary landing aid to be installed will be a permanent TACAN located in close proximity to the runway.

3.4.2.3.6 Meterological. Meteorological measurements and reports will be required.

3.4.2.3.7 Medical. Emergency medical support shall be provided.

3.4.2.3.8 Orbiter/Ground Communications. A UHF air traffic control voice capability will be provided for voice comm between the Tower and the MCC.

3.4.3 <u>Contingency Landing Stations</u>.

3.4.3.1 Performance Characteristics. The Contingency Landing Stations will be acceptable premission selected sites providing for crew and passenger survival with no Orbiter vehicle unique support specifically implemented.

3.4.3.2 Design Characteristics.

3.4.3.2.1 Runway. The selected existing runway will be at least equivalent to the Orbiter design runway (Sect. 3.3.1.1.8) with a length of 12,100 feet.

3.4.3.2.2 Landing Aids. The primary landing aid will be an existing TACAN.

3.4.3.2.3 Orbiter/Ground Communications. The existing UHF air traffic control voice capability will be required.

3.4.3.3 (Deleted).

3.4.4 <u>Orbiter Processing Facility</u>.

3.4.4.1 Performance Characteristics. A capability shall be provided to perform the following functions:

a. Orbiter and Payload Safing and Deservicing

b. Hypergolics Pod Removal and Installation

c. Payload Removal and Installation and Interface Verification

d. Orbiter Maintenance including TPS Refurbishment

e. SSME on Orbiter Maintenance and Engine Changeout

f. Validation of Orbiter and Payload communications Interfaces with the Support Network including an RF interface (such as Ku-Band, S-Band) with the network TDRS test station - MILA.

g. Verify Orbiter Systems for Flight

h. Installation/Removal of Ferry Kit

3.4.4.2 Design Characteristics.

3.4.4.2.1 Control/Monitor Equipment. Ground support control and monitor equipment provided by the LPS shall interface with onboard serial digital control and data management subsystem and supporting GSE. This equipment may be controlled from the LCC.

3.4.4.2.2 Environment Protection. The Orbiter Processing Facility shall be of hangar type construction and shall provide only basic protection from the elements. Special provisions for environmental protection for payload removal and installation shall be defined in 3.6.12.2 and shall provide class 100,000 conditioned air to the crew compartment.

3.4.4.3 Interface Characteristics. The Orbiter Processing Facility shall interface with the Orbiter as defined in Orbiter/Processing Station ICD 2-1A002.

3.4.5 Shuttle Vehicle Assembly and Checkout Stations

3.4.5.1 Performance Characteristics. Two Shuttle vehicle assembly and checkout stations shall be provided in the Vehicle Assembly Building (VAB) to perform the following functions:

a. Stacking and alignment of LRB on Mobile Launcher Platform (MLP) *

b. Erection and mating of ET to LRB *

c. Erection and mating of Orbiter to ET

d. Element integration and interface checkout and verification. *

e. Checkout of Space Shuttle Flight Vehicle

f. Installation of pyrotechnics

g. Provide lightning protection system in accordance with NSTS 07636

h. LRB engine changeout if necessary. *

3.4.5.2 Design Characteristics.

3.4.5.2.1 Control/Monitor Equipment. Ground support control and monitor equipment provided by the LPS shall interface with onboard serial digital control and data management subsystems and supporting GSE. This equipment may be controlled from the LCC.

3.4.5.2.2 Access Platforms. The station shall be provided with platforms in support of LRB, ET, and Orbiter Vehicle handling, mating, servicing, and checkout. *

3.4.5.2.3 Roadway. A roadway capable of supporting crawler/transporter with the MLP and total assembled Shuttle vehicle shall be provided between the Shuttle vehicle assembly and checkout station and the launch pad station.

3.4.5.2.4 Interface Characteristics. The Shuttle Vehicle Assembly and Checkout Station shall interface with the Shuttle Vehicle as defined in (TBD) *

3.4.6 Launch Pad Station.

3.4.6.1 Performance Characteristics. Two launch pads and mobile launch platforms shall be provided to perform the following functions:

a. Support the fully assembled flight vehicle in the vertical attitude for transportation from the vehicle assembly station to the launch station

b. Prelaunch checkout

c. Vehicle and payload servicing

d. Countdown

e. Personnel ingress

f. Payload removal and installation

g. Prelaunch escape for flight crew, passengers, and ground crew from flight vehicle interface to a safe area in 90 seconds

h. (Deleted)

i. Validation of Orbiter and Payload communications interfaces with the support network including an RF interface (such as Ku-Band, S-Band) with the network TDRS test station - MILA

j. SSME and/or LRB engine removal and installation *

k. Contingency access to the Orbiter, ET and LRB TPS and the Orbiter/ET attach fittings and umbilical connectors will be provided for post-FRF inspection and repair.

3.4.6.2 Design Characteristics.

3.4.6.2.1 Control/Monitor Equipment. Ground support control and monitor equipment provided by the LPS shall interface with onboard serial digital control and data management subsystem and supporting GSE. This equipment shall be controlled from the LCC and shall also interface with the onboard RF system.

3.4.6.2.1.1 LRB Engine Shutdown. Ground support equipment shall provide * the capability to command shutdown of all or selected LRB engines prior to liftoff.

3.4.6.2.2 Holddown. A holddown capability shall be provided to hold the flight vehicle on the mobile launch platform during thrust buildup to 100 percent RPL. The holddown subsystem shall withstand the effects of ground winds, vehicle * dynamics, thrust vector alignment and thrust vector excursions.

3.4.6.2.3 Service Tower. The service tower shall be provided with:

a. Elevators

b. Crew access and emergency egress

c. Vehicle reactant storage

d. Crane

e. Lightning protection in accordance with NSTS 07636

f. Umbilical support for ET G02 venting and GH2 vent/GUCP.

g. Umbilical support for safe LRB oxidizer venting and fuel venting *

3.4.6.2.4 Power Sources. The launch pad facility shall be provided with an emergency safing power source in addition to the primary and secondary power source.

3.4.6.2.5 Gas Supply. A gas storage area and associated lines shall be provided at the launch station to provide gases for the Shuttle vehicle.

3.4.6.2.6 Propellant/Reactant Loading. Cryogenic propellant/reactant loading capability of the Shuttle ground system shall be as follows:

a. Simultaneous or sequential LO_2/LH_2 main propellant and booster propellant loading or drain. *

b. Concurrent and sequential flight vehicle main propulsion, booster and payload propellant loading. *

c. Emergency drain capability shall not be precluded through the normal fill and drain system.

d. Main propellant and booster propellant fast fill loading shall not require onboard personnel support, shall occur prior to crew and passenger ingress, and shall be completed within approximately 114 minutes. *

e. Reactant loading and closeout shall be completed prior to "standby" (see 3.2.1.2.2) and shall not require onboard personnel support.

3.4.6.2.7 Storable Propellant Loading. Storable propellant (hypergolic) servicing, including connection and disconnection, shall be accomplished in 13 hours. Emergency drain capability shall be provided.

3.4.6.2.8 Venting. Venting capability and disposal of hazardous vapors shall be provided to satisfy all Shuttle vehicle and payload requirements.

3.4.6.2.9 Personnel Loading. Transfer from the blast danger area roadblock to the Orbiter, Flight personnel and passengers ingress, cabin closeout and prelaunch checks, including terminal count, must be accomplished in 60 minutes.

3.4.6.2.10 Acoustic Deflection/Suppression. Plume deflection and water injection shall be provided to minimize the acoustic environment on the Shuttle vehicle, payloads, and ground facilities. Plume heating and water injection shall not impact the TPS design requirement and there shall be no direct water impingement on the LRB or SSME nozzles. No water shall be deposited on the SSME or LRB main combustion chamber internal surface above the plane of the nozzle throat. *

3.4.6.2.11 Removal and Installation. The launch pad facility shall provide the capability to remove and install the SSMEs and LRB engines. *

3.4.6.2.12 Payload Contamination Control. Purging and atmospheric control of the payload bay independent of the Orbiter vehicle internal structure shall be provided by GSE with the payload bay doors opened or closed.

3.4.6.2.13 Payload Coolant. Demineralized, deionized water shall be provided by GSE to the Orbiter T-0 umbilical for ground cooling of payloads.

3.4.6.2.14 Payload Changeout Room. The Payload Changeout Room (PCR) shall have provisions to support vehicle and payload activities.

3.4.6.2.14.1 Vehicle Support Provisions

a. Vehicle reactant loading system and associated access

b. Vehicle storable propellant distribution system and associated access

c. Provide for access to the Orbiter preflight umbilical and the mid-body access door from the PCR when mated to the Orbiter

3.4.6.2.14.2 Payload Support. Provisions shall be made to accommodate payload upper stage mating, interface verification, servicing, systems checkout, installation and/or removal and associated access as follows:

a. Changeout - The PGHM shall provide capability for on-pad changeout of payloads, including replacement of removed payloads by dissimilar payloads. Vertical installation/ removal reconfiguration capability shall be provided for the following Orbiter payload bay flight kits:

    1. Payload structural attachments

    2. Standard Mixed Cargo Harness (SMCH) and Spacelab harness

    3. Mid and aft fuselage ballast

    4. OMS Delta V propulsion module (requires 3-point longeron attachment)

    5. T-4 hours payload umbilical (requires payload removal above umbilical)

    6. Rescue

b. Payload PCR Occupancy - PCR design and operations shall not preclude payload occupancy of the PCR for PCR reconfiguration, checkout, and servicing operations immediately after pad safing operations following launch. Payload checkout will be completed prior to the arrival of the Shuttle at the pad.

Facility services shall be provided (e.g., power, contamination control, etc.) from payload installation into the PCR until payload installation in the Orbiter and payload bay doors are closed, or payload removal from the PCR.

Alternate or redundant services shall be provided to assure a return to a safe configuration and to maintain environmental control.

c. Environmental and Contamination Control - Contamination control for the PCR shall be defined in Paragraph 3.6.12.2.4. Environmental control for the PCR shall be maintained at 70 ± 5 deg F and 30-50% relative humidity. The PCR shall not preclude the installation of a payload supplied local enclosure. The enclosure shall be purged with the localized air conditioning as defined in subparagraph below.

   Provide a localized air conditioning system at the 20 and 40 foot work platform levels on the east (outboard) side of the PCR. Flexible ducting between the outlets and the payload will be provided by the user. Requirements are:

   1. Temperature - Adjustable between 52° to 75°F min/max range including ± 3°F at a given setting.

   2. Humidity - Within 30 - 50% RH

   3. Total Flow - Adjustable between 0 - 250 lb/min max

   4. Contamination Control - Same as PCR air (reference Paragraph 3.6.12.2).

d. Access - With the payload installed in PCR and with the payload handling mechanism in the retracted position, the PCR shall:

   1. Provide five fixed platforms, in addition to the base floor for payload support operations. Each platform shall provide space of approximately 600 square feet which shall be allocated for payload-related equipment use. This area shall be designed for 100 lb/ft^2. Platform surfaces and joints shall be designed to minimize the fall-through of contamination, debris, and small hardware items.

   2. Utilizing the platforms specified in d.1 above (except for small areas blanked by payload attachment fittings), provide a capability for 360° access around the longitudinal axis of a 60-foot cylindrical payload 5 to 15 feet in diameter using extensible or insert platforms designed for 50 lb/ft^2. A two-foot platform width in the horizontal plane shall be a design goal for a 15-foot diameter payload. Insert platforms shall minimize fall-through of contamination, debris, and small hardware items.

      With the payload installed in the Orbiter payload bay, the PCR shall provide:

1. Personnel and equipment access platforms to all Orbiter/payload accommodation interfaces with payload installed in the payload bay, with live load capability of 50 lbs/ft^2.

2. Personnel and equipment access to exposed surface of payload installed in the payload bay throughout entire payload bay length when bay doors are open and PGHM extended mold line for unique servicing and adjustments of payload elements. These access provisions shall include five fixed PGHM platforms with attachable inserts, and a live load capability of 100 lbs/ft^2 for fixed and 50 lbs/ft^2 for attached insert platforms.

   Access of two-feet below payload and six-feet above a 60-foot long payload shall not be precluded by structural dimensions of the PCR payload area.

   Provisions for access to installation/removal of mid-fuselage Orbiter LRUs, payload liner, keel fittings, bridge fittings, and mission kits, when the PCR is mated to the Orbiter and the payload bay doors are open and payload not in Orbiter or PCR.

   Provisions for access to the Orbiter payload bay doors, interior (radiators) when payload bay doors are open, and exterior when payload bay doors are closed.

e. Handling

1. Provisions for handling individual payloads and multiple payloads (up to five) weighing up to 65,000 pounds total for mating/demating a satellite to an upper stage installed in the PCR, and for installation or removal of integrated payload(s) to or from the Orbiter payload bay shall be provided.

2. Payload Component Handling Equipment Hoist System - A hoist system shall be provided for lifting payload components and handling equipment off payload segments installed on the PGHM and translation to the fixed work platforms. The maximum single item weight is 1000 lbs.

3. Devices shall be provided for lifting GSE and payload elements up to 8,000 lbs. to and from various platform levels within the PCR, without retraction of the PCR.

f. Ingress/Egress

1. Provide an airlock for ingress/egress of payload operating personnel, payload-related ground support equipment and payload elements. The airlock shall be sized to accommodate equipment sizes up to a maximum envelope of 6'W X 8'H, and weight not to exceed 8,000 lbs. and a live load of 100 lbs/ft^2. An anteroom of

sufficient size shall be provided for use in conjunction with the airlock to accommodate a security guard station, clean clothing change, clothing storage, shoe cleaning equipment, and a personnel air shower. The anteroom shall be fully enclosed and all utilities provided to accommodate safe cleanroom operations.

2. The capability for emergency egress shall be provided on each side of each primary work level, and shall include internal passageways, exit doors, and stairways that are external to the PCR enclosure.

3. An equipment storage room shall be located adjacent to the anteroom to store, in a clean environment, the tools and equipment used in the payload changeout room. Provisions shall be made for GSE carts used in vehicle processing.

g. Communications/Data

Provide communications from the PCR to the appropriate ground station (voice, landline, coax, direct RF) as follows:

1. Landlines

(a) Voice Communication - Payload voice communication capability will be provided by the NASA Operational Intercom System (OIS) within KSC and to specific locations within CCAFS.

(b) Data Lines - Payload data lines shall be provided from the launch complex to specific locations within KSC and to the CCAFS interface.

(c) S-Band Pick-Up Antenna - Provide S-Band pick-up antenna to be used during pad RF open-loop payload interrogator (PI)/S-Band interface test; cabling to connect antenna to facility PI interface at the PCR. The pick-up antenna shall be capable of supporting various S-Band antenna locations for different payloads.

2. RF Communications - Provide an open-loop RF communication capability for the payload(s) to communicate within line-of- sight from the PCR to the payload facilities.

(a) The RF links to be provided are:

Frequency Band

S-Band (SGLS)

X-Band

L-Band

UHF

(b) Data Relay Antennas - Structural supports for antennas for the above frequency bands shall be provided on the exterior of the PCR. Antennas as required will be provided by users. Cable trays, waveguide supports, penetration plates and personnel access platforms shall be provided for installation of equipment.

3. Telephone - Telephones shall be provided at all internal PCR levels and at specified critical locations to provide the capability for "on" and "off" site communications. Each telephone handset shall have a press-to-talk capability or an equivalent confidence device.

4. Timing - IRIG B timing signals shall be provided at all interior PCR platform levels and at the pad surface park site for trailerized AGE vans.

h. Electrical Power

1. Provide electrical power complying to MIL-STD-1542, for the following services at both sides of all interior PCR platform levels. Connected GSE load will not exceed 100,000 BTU/hr. into the PCR.

(a) 120/208 VAC, 3 phase, 60 Hz at, 36 KVA each

(b) 120 VAC, single phase, 60 Hz at, 4.8 KVA each

(c) 28 VDC, 30A

A single-point emergency manual cutoff, one each for AC and DC, shall be provided at each major work level for the electrical power supplied at the level.

2. The following electrical power outlets per MIL-STD-1542 shall be provided on the pad surface in the area of the PCR:

(a) 480 VAC, 3 phase, 60 Hz at, 104 KVA

(b) 230 VAC, single phase, 60 Hz at 3 KVA

(c) 208 VAC, single phase, 60 Hz at 12.5 KVA

(d) 120 VAC, single phase, 60 Hz at 21 KVA

(e) 120/208 VAC, 3 phase, 60 Hz at 36 KVA each

i. Propellant/Consumable Servicing

1. Provide for payload storable propellant loading of less than 1200 lbs. mass with payload provided GSE and launch facility provided pressurization vent, and drain system.

Propellant servicing of greater than 1200 lbs. mass shall be accomplished by launch facilities through an Orbiter * or LRB umbilical.

2. Provide the capability for supporting payload consumable loading/unloading, pressurization venting, and draining. Payload consumable handling will be accomplished by payload GSE.

j. Security

1. Personnel Access - Personnel access capability into and out of the PCR shall be provided at an airlock for normal operations (excludes emergency egress). Access provisions shall allow up to 15 support personnel to enter/leave the room within a 15 minute period.

2. Visual and Aural Access - For DOD payloads neither the payloads nor the payload GSE shall be visible from outside the PCR enclosure after installation in the PCR. The DOD will provide the controls and/or equipment required to prevent visibility by unauthorized personnel when such personnel are required to be inside the PCR enclosure. Aural communications generated inside the secure PCR area shall be attenuated such that communication security of mission data shall not be compromised.

k. Payload Cleaning - A built-in vacuum system for cleaning payload elements and support equipment shall be provided with inlets in the airlock, anteroom and both sides of each interior PCR work platform level.

l. RF Shielding

1. External Radiations - The PRC, with PCR doors closed, shall provide attenuation of the local external (to the PCR) RF environment such that the extraneous RF environment in the vicinity of the DOD payload does not exceed 1 volt/meter over the frequency range from 15 kHz up to 30 GHz.

2. Communication Security - The PCR, with the PCR doors closed shall provide attenuation of payload-generated emmisions such that communication security of mission data will not be compromised.

m. Pneumatics

1. Work Platform Pneumatic Supply - A GN_2 and He manifold shall be provided to all work platform levels on the west (pivot point) side of the PCR. Outlets shall be capable of providing 3000 + 100 (psig) per NASA Specification NSTS SE-S-0073B.

2. Special Purity GN_2 Supply Line - A stainless steel line capable of 4 $ft^3/hr.$ at 30 psig shall be provided from the launch pad to an outlet between the 30 and 50 foot work platform levels on the west side (pivot point) of the PCR. The user will provide controls at the outlet to regulate the supply. A special purity GN_2 supply trailer will be provided by the DOD at the pad surface.

n. Utility Air

1. A compressed air source within the PCR (up to 120 psig) shall be provided with outlets appropriately positioned on the major work platform levels.

2. A compressed air source (up to 120 psig) shall be provided at the pad surface for use under the canister installed on the retracted PCR, and the adjacent area where payloads will be off-loaded from transporters.

o. Support Trailers

1. An area on the launch pad adjacent to the PCR hinge column shall be provided for parking up to three 10' X 50' trailers containing payload checkout equipment.

2. An area adjacent to the ground inlet of the special purity GN_2 supply line shall be provided for parking an approximately 8' X 40' DOD GN2 trailer.

p. Payload Canister - A payload canister shall be provided to transport and install payloads up to the 65,000 lbs. and maximum 15' diameter 60' long configuration.

1. Environmental Control - Payload environmental requirements include the following:

(a) Temperature - 70 + 5°F

(b) Humidity - within 30 - 50% R.H.

(c) Cleanliness - per Paragraph 3.6.12.2.4

(d) Shock Vibration - The sum of static and dynamic loadings sustained by the payload during handling and transportation shall be controlled to be significantly below the design flight loads of the payloads. These design flight loads are not to be expected to be below the following:

axial 5.0g (Compression)
-2.0g

lateral $\pm 2.5g$

angular   0.02g/in

2. Gas Service - Payloads will require the following gas service in the canister:

   (a) Special Purity GN_2 for continuous purge of payload elements (30 psi at 4 $ft^3/hr.$). A mount for K-bottle and a bulkhead fitting are required on the canister. Bottle regulator and flexible lines will be provided by the user.

3. Electrical Power - Payload electrical power requirements during transportation include the following:

   (a) 115/120 Vac, 10/, 60 Hz at 1.8 KVA

   (b) Space/mounting provisions for payload power supply unit (2' wide X 2' high X 2' long)

4. RF attenuation - The canister shall provide sufficient RF attenuation capability to prevent internal power levels * received from local sources from exceeding values established by ESMCR-127-1.

3.4.6.2.15 Orbiter/SSME Fire Protection. A water spray system shall be provided to protect all surfaces of the Orbiter aft fuselage from a SSME post-shutdown potential hydrogen fire. The water spray system provides water spray coverage of the Orbiter/ET 17" disconnects also.

3.4.6.2.16 Disposition of Unburned Hydrogen in SSME Exhaust. The facility shall preclude the buildup of a quantity of hydrogen mixture that could detonate and cause Orbiter structural damage under the following situations:

a. On-pad SSME Startup Sequence

b. Nominal On-pad Engine Firing

c. FRF Shutdown

d. On-pad Abort Shutdown

3.4.6.2.17 LRB Conditioning and Instrumentation Umbilicals shall provide functions prescribed in Paragraph 3.3.2.3.8. These umbilicals shall accommodate *
the thermal environment identified in (TBD) *
acoustic environment identified in (TBD) *
ignition overpressure environment identified (TBD)

*
3.4.6.2.18 LRB Fire Protection (TBD)

3.4.6.3 Interface Characteristics. The Launch Pad Station shall interface with the Shuttle Vehicle as defined in the Shuttle System/Launch Pad and MLP ICD 2-0A002.

3.4.7 External Tank Processing and Storage Station

3.4.7.1 Performance Characteristics. An area shall be provided in the VAB for receipt. checkout and storage for four external tanks, their components, and ground support equipment.

3.4.7.2 Design Characteristics.

3.4.7.2.1 Lifting and Handling. Provisions shall be made for moving and handling of the tank within the storage areas and for movement of the tank to the vehicle assembly area.

3.4.7.2.2 Environmental Protection. The processing and storage station shall provide only basic protection from the elements.

3.4.7.2.3 Access. The station shall have access provisions for the external tank in either the vertical or horizontal attitudes, including internal and external access to the tank.

3.4.7.3 Interface Characteristics. The External Tank Procsssing and Storage Station shall interface with the External Tank as defined in External Tank/Receiving, Storage, and Checkout Station ICD 2-2A001.

3.4.8 Liquid Rocket Booster Processing and Storage Station *

3.4.8.1 Performance Characteristics. An area shall be provided for the receipt, handling, inspection and storage of LRB subassemblies, and for the assembly and handling of the LRB Assemblies prior to their movement to the VAB for final * LRB assembly and checkout. Facilities for alignment, dimension, weight, and CG measurements shall also be provided.

3.4.8.2 Design Characteristics.

3.4.8.2.1 Lifting, Handling and Transporting. Provisions shall be made for movement of the LRB from the storage area to the assembly area, lifting and * handling of the LRB within the assembly area, and movement of the LRB from the work area to the vehicle assembly area.

3.4.8.2.2 Hazardous Operations. The facility shall be located such that hazardous operations will not impact other unrelated site activites.

3.4.8.2.3 Environmental Protection. The assembly area shall be located within an environmentally controlled enclosure. The storage area shall be located * within an environmentally protected area or provisions made for protecting the LRB from the elements.

3.4.8.2.4 (Deleted)

3.4.8.3 Interface Characteristics. The Liquid Rocket Booster Processing Station shall interface with the Liquid Rocket Booster as defined in the * LRB/Receiving and Checkout Station (TBD)

3.4.9 LRB Retrieval and Disassembly Station/Recovery System *

3.4.9.1 Performance Characteristics. Capabilities shall be provided for the retrieval, return, disassembly, cleaning, preservation and shipment of the expended LRBs. *

3.4.9.2 Design Characteristics. Provisions shall be made for retrieval of the LRBs at splashdown and handling of the expended LRBs from retrieval through the station to shipment. *

3.4.9.3 Interface Characteristics. The Liquid Rocket Booster Retrieval and Disassembly Station shall interface with the LRBs as defined in the LRB/Retrieval Station (TBD) *

3.4.9.4 The LRB Recovery System shall be capable of timely deployment and support of the recovery crew and equipment to the recovery site(s). *

3.4.9.5 The LRB Recovery System shall have the capability to: *

a. Safe LRB System when recovered. *

b. Transport the recovered flight hardware to the launch site. *

c. Transfer LRB hardware from the recovery equipment to the appropriate refurbishment station (Parachute, 3.4.10, LRB, 3.4.17) *

3.4.10 <u>Parachute Refurbishment Station</u>

3.4.10.1 Performance Characteristics. An area and support equipment shall be provided for the cleaning, drying, inspection, repair, repacking, and storage of the LRB recovery system parachutes. *

3.4.10.2 Design Characteristics. The station shall be environmentally controlled.

3.4.10.3 Interface Characteristics. Interface provision shall be compatible with the requirements of Paragraph 3.4.10.1.

3.4.11 <u>Hypergolic Maintenance and Checkout Station</u>.

3.4.11.1 Performance Characteristics. A facility shall be provided to perform the following functions off-line for the vehicle systems designated below:

3.4.11.1.1 Orbiter FRCS, Orbiter APS, and Orbiter PBK.

a. Service, drain, flush, and purge

b. Leak and functional test

c. Refurbishment and maintenance

d. Storage and handling

3.4.11.1.2 Orbiter APU

a. Flush and purge

b. Handling and preparation for shipment

3.4.11.1.3 Liquid Rocket Booster APU. *

a. Service, drain, and purge.

b. Handling and hot fire. Hot fire requirements apply through the DDT&E phase only. Operational period requirements for hot firing Liquid Rocket Booster HPU will be determine by completion of OFT. *

3.4.11.2 Design Characteristics. The faiclity shall be located such that hazardous operations will not impact other unrelated site activities and be designed so that operations and maintenance can be performed simultaneously on different modules with no operational impact.

3.4.11.3 Interface Characteristics. The Hypergolic Maintenance and Checkout Station shall interface with Orbiter modules as defined in Orbiter/Hypergolic Station ICD 2-1A003.

3.4.12 Engine Maintenance Station. *

3.4.12.1 Performance Characteristics. A capbility shall be provided to perform the following functions for SSMEs and LRB engines:

a. Inspection

b. Repair

c. Checkout

d. Storage

3.4.12.2 Design Characteristics. The station shall be located within an environmentally controlled enclosure.

3.4.12.3 Interface Characteristics. Interface provisions shall be compatible with the requirements of Paragraph 3.4.12.1.

3.4.13 (Deleted).

3.4.13.1 (Deleted).

3.4.13.2 (Deleted).

3.4.13.3 (Deleted).

3.4.14 Flight Crew System Station.

3.4.14.1 Performance Characteristics. A capability shall be provided for the storage, repair, maintenance and servicing of flight crew system equipment. This station will also be utilized by Flight Crews for purposes of Bench Reviews/Familiarization of Flight Crew System equipment.

3.4.14.2 Design Characteristics. The station shall be located within an environmentally controlled enclosure.

3.4.14.3 Interface Characteristics. Interface provisions shall be compatible with the requirements of Paragraph 3.4.14.1.

3.4.15 <u>LRU Maintenance Station</u>.

3.4.15.1 Performance Characteristics. Shop and laboratory capability shall be provided for the maintenance, repair, test, analysis, acceptance and packaging of designated Shuttle system LRUs.

3.4.15.2 Design Characteristics. The station shall be located within an environmentally controlled enclosure.

3.4.15.3 Interface Characteristics. Interface provisions shall be compatible with requirements of Paragraph 3.4.15.1.

3.4.16 <u>Launch Process System Station</u>.

3.4.16.1 Performance Characteristics. A Launch Processing System (LPS) shall be provided to perform monitor, control, data processing and display in support of maintenance, test, checkout, launch control, and operational management of Shuttle vehicle, payloads and ground systems involved in launch site ground turnaround operations. The LPS shall have the capability to exchange information with other data systems. Maximum use of the onboard capability for checkout of flight systems shall be a goal where cost and turnaround considerations warrant. This capability shall be augmented by the ground systems where necessary.

3.4.16.2 Design Characteristics.

3.4.16.2.1 General. The LPS shall consist of an integrated network of computers, data links, displays, controls, hardware interface devices, and computer software designed to control and monitor flight systems, payloads, and those GSE and facilities utilized for direct support of vehicle activities.

3.4.16.2.2 Automated or Manual Capability. The LPS shall provide for automatic and manual sequencing and control with Operator override capability.

3.4.16.2.3 Exception and Continuous Monitoring. Exception and continuous monitoring capability is required. A capability shall exist to select the specific measurements to be monitored and to revise the limits associated with exception monitoring.

3.4.16.2.4 (Deleted).

3.4.16.2.5 Fault Isolation. The capability to perform fault isolation to an LRU or group of LRUs within the flight or ground systems shall be provided by the combined onboard and ground system (Reference NSTS 07700-10-MVP-01, Paragraph 3.6.2). The LPS shall augment the onboard fault isolation programs to provide fault isolation for flight vehicle systems (Reference Paragraph 3.3.1.2.3.8).

3.4.16.2.6 Uplink capability. Capability shall be provided to initiate uplink commands in a format compatible with the Shuttle data processing and/or uplink command system as defined by ICD-2-0A003, Flight Vehicile/LPS Computational Systems Interfaces. (Both hardware and RF capability shall exist.)

3.4.16.2.7 Real Time Data Display. The operator engineer shall be provided the capability to access Shuttle, payloads, and ground systems test data in realtime for display as required to support ground turnaround operation.

3.4.16.2.8 Test Data Recording. A capability shall be provided to record all raw test data (downlink) prior to any preprocessing and all commands transmitted (uplink).

3.4.16.2.9 Historical Test Data Retrieval and Display. A capability shall be provided to retrieve and display historical test data.

3.4.16.2.10 Interactive Data Analysis. Provision shall be made to access recorded ground turnaround operations data via remote terminals.

3.4.16.2.11 Engineering Data File (EDF). The LPS shall include an Engineering Data File which will make available the operational management information necessary to control, manage, and status Shuttle processing. Specific data systems to be maintained in the EDF shall be as specified in NSTS 07700, Volume V.

3.4.16.2.12 Remote Area Terminal. The LPS shall have the capability to interface with remote area terminals.

3.4.16.2.13 LPS Software System. The LPS software system shall provide a medium by which the test engineer can effectively and efficiently communicate with the test article through the LPS computer system. Automated checkout programs shall be operable in the operator intervention mode as well as automatic mode. The software system must be capable of supporting the functions allocated to the LPS by NSTS 07700, Volume XVIII, Books 1 and 2.

3.4.16.2.14 Post Test/Launch/Mission Data Reduction and Evaluation. The LPS shall have a capability to provide assessment of system anomalies encountered during ground turnaround operations or launch. In addition, the capability for assessment of selected flight data is required to support shuttle maintenance requirements.

3.4.16.3 Interface Characteristics. The LPS shall interfaced with other Shuttle elements and data systems as defined in Volumes V and XVIII of NSTS 07700, ICD 2-0A003 and Computer Program Development Specification SS-P-0002-150.

3.4.17 <u>LRB Refurbishment and Subassembly Station</u> *

3.4.17.1 Performance Characteristics. An area shall be provided for the receipt and storage of LRB subassemblies, from the Vendor or from the LRB Retrieval and Disassembly Station, and for their handling, inspection, refurbishment, assembly, and verification for flight prior to their movement * to the LRB Processing and Storage Station at KSC or VLS, for mating with the LRB Assemblies. Provisions to access the interior and exterior of the LRB shall be provided.

3.4.17.2 Design Characteristics.

3.4.17.2.1 Lifting, Handling and Transporting. Provisions shall be made for movement of the LRB subassemblies from the storage area, to the refurbishment, assembly, and verification areas, lifting and handling within these areas, * and movement to the LRB Processing and Storage Station at KSC or VLS.

3.4.17.2.2 Hazardous Operations. The facility shall be located such that hazardous operations will not impact other unrelated site activities.

3.4.17.2.3 Environmental Protection. The storage, refurbishment, assembly, and verification areas shall be environmentally controlled.

*

3.4.17.2.4 Ground System. (Deleted)

3.4.17.3 Interface Characteristics. The LRB Refurbishment and Subassembly Station shall interface with the LRB subassemblies as defined in the LRB/Receiv- * ing and Checkout Station. (TBD)

3.4.18 <u>Orbiter and Carrier Aircraft/Mate Demate Station</u>.

3.4.18.1 Performance Characteristics. The capability shall be provided to mate and demate the Space Shuttle Orbiter and the Shuttle Carrier Aircraft (SCA), to support the ferry mission.

3.4.18.2 Design Characteristics.

3.4.18.2.1 Mate/Demate Device. The Mate/Demate Device (MDD), shall provide the capability to perform the physical mating. The demate operation shall be essentially the same, except done in reverse sequence.

3.4.18.3 Interface Characteristics. The Orbiter and Carrier Aircraft/Mate Demate Station shall interface with the Orbiter and Carrier Aircraft as defined in ICD 2-1D004.

3.4.19 <u>Cargo Interface Verification Equipment</u>. The purpose of this equipment is to provide maximum assurance of Orbiter/cargo compatibility prior to installation of the cargo into the Orbiter payload bay.

3.4.19.1 Performance Characteristics. Equipment shall be provided to simulate the physical and functional interfaces between the Orbiter and cargo. The equipment shall provide capability to perform the following functions:

a. Aid the verification of cargo form, fit, and function within the Orbiter cargo bay.

b. Support up to 5 cargo elements with a maximum weight of 65,000 lbs. in either the horizontal or vertical mode.

c. The structural, mechanical, and electrical systems shall be designed for use as individual pieces of equipment to support testing and for combined use integration test requirements.

3.4.19.2 Design Characteristics.

3.4.19.2.1 The major structural elements are:

a. Mid-body structure mechanical interfaces shall simulate a 15-feet wide by 60-feet long Orbiter bay. Provision shall be provided for installing cargo attach fittings in the longeron and keel area to accommodate the Orbiter 3.933-inch vernier concept. Dimensional tolerance for the cargo/Orbiter interface locations shall be capable of control to one-half the Orbiter fabrication tolerance for those locations (Reference Orbiter tolerance drawing VL 70-004260).

b. Aft crew station support structure and the MS/PS/OOS consoles.

c. Xo 576 bulkhead assembly

d. Xo 1307 bulkhead assembly

e. Payload wire trays

f. Preflight umbilical panel provisions

g. RMS and door actuator critical interference envelopes

h. Floor supports

i. Primary longeron fitting-nondeployable payload

j. Stabilizing longeron fitting-nondeployable payload

k. Auxiliary keel fitting

l. Xo 693 power interface panel

m. Xo 576 Airlock interface/Xo 660 tunnel interface structural provisions. (Provisions of Airlock and tunnel interfaces is not a Space Shuttle DDT&E requirement.)

3.4.19.2.1.1 Cleanliness. The equipment shall be designed to be compatible with and facilitate the maintenance of a class 100,000 environment as specified in Federal Standard 209B.

3.4.19.2.2 The Interface verification equipment shall demonstrate Orbiter to cargo/electrical interface compatibility by simulating the Orbiter interface within specification tolerances. The electrical system shall perform the following functions:

a. Thruput digital command/data, discretes and analog signals from the cargo to the cargo support GSE ("bent pipe").

b. Provide encoded digital commands and discrete signals as required to the cargo subsystems.

c. Perform quantitative data processing of selected analog, discrete and digital data. This includes:

1. Simulating the cargo related data handling capabilities of the Orbiter Communications and Data Handling (C&DH) system.

2. Performing functional testing of the cargo as required to verify Orbiter interfaces.

3. Simulating the Flight Computer Operating System (FCOS) response to payload data, (timing, etc.).

4. Simulating all cargo related Orbiter data outputs from the FCOS and interleaving of Orbiter and payload data.

d. Provide a test measurement system for monitoring payload interface signal characteristics.

e. Provide a simulation of Orbiter bus power for cargo subsystems.

f. Provide a source of AC and DC to payload Display and Control (D&C) equipment in the payload station.

g. Provide a closed loop simulation of the Orbiter/cargo RF link.

3.4.19.2.2.1 Electromatnetic Compatibility. The design objective shall be to minimize the generation of and susceptibility to electromagnetic interference.

3.4.19.2.3 It shall be a design goal that the interface verification equipment can be disassembled, transported to a new site, reassembled and verified for operations.

3.4.19.2.4 It shall be a design goal that the electrical system be capable of stand-alone operation, provide flexibility in performance and operation over the Orbiter to cargo interface specification ranges, and provide for growth of additional requirements.

3.4.20 <u>SRM Storage Stations(s)</u>. (Deleted) *

3.4.20.1 Performance Characteristics (Deleted) *

3.4.20.2 Design Characteristics (Deleted) *

3.4.20.2.1 Lifting, Handling and Transporting (Deleted) *

3.4.20.2.2 Hazardous Operations. (Deleted) *

3.4.20.2.3 Environmental Protection. (Deleted) *

3.4.20.2.4 Ground System. (Deleted) *

3.4.20.3 Interface Characteristics. (Deleted) *

3.4.21 <u>External Tank Storage Station(s)</u>.

3.4.21.1 Performance Characteristics. An External Tank storage capability shall be incrementally developed to support the program mission model. Minimum capability shall be 6 External Tanks at the project contractor's facility.

3.4.21.2 Design Characteristics.

3.4.21.2.1 Lifting and Handling. Provisions shall be made for the moving and handling of the External Tanks within the storage station.

3.4.21.2.2 Environmental Protection. The storage area shall protect the External Tanks from the elements as follows: No rain or snow; temperature, no lower than 11°F; and humidity - No condensation on the hardware.

3.4.21.2.3 Ground System. A grounding system shall be provided within the storage area to prevent static charge buildup on the External Tank.

3.4.21.2.4 Access. The storage station shall have access provisions for the External Tank in the horizontal attitude, including internal and external access to the tank.

3.4.22 Flight Operations *

3.4.22.1 Flight Operations consists of the support functions provided by the Mission Control Center (MCC) and the Huntsville Operations and Support Center (HOSC). These facilities shall function as part of integrated network as defined in Launch Processing System Station (3.4.16) *

3.4.22.1.1 Flight Operations shall have the ability to monitor shuttle propellant levels temperatures and pressures. *

3.4.22.1.2 Flight Operations shall have the ability to monitor shuttle engine parameters. *

3.4.22.1.3 Flight Operations shall have the ability to monitor shuttle engine ignition, startup, and shutdown sequences. *

3.4.22.1.4 Flight Operations shall have the ability to monitor LRB separation sequence, LRB deceleration sequence, and LRB descent and impact location(s). *

3.4.22.1.5 Flight Operations shall have the capability to monitor and verify:

a. Engine TVC prior to liftoff.

b. Range Safety System (RSS) communication and data links prior to liftoff *

c. LRB engine shutdown and separation *

d. LRB deceleration system deployment *

e. Shuttle abort sequence parameters.

3.5 OPERABILITY.

3.5.1 Reliability. Shuttle System reliability shall be in accordance with NHB 5300.4(1D-2). Any Deviation/Waiver of reliability requirements shall be in accordance with NSTS 22206.

3.5.1.1 Flight System Reliability.

3.5.1.1.1 Flight Vehicle Subsystem Reliability. The redundancy requirements for all flight vehicle subsystems (except primary structure, thermal protection system, and pressure vessels) shall be established on an individual subsystem basis, but shall not be less than fail safe during all mission phases including intact aborts with the exception of the subsystem causing the abort.

In addition to Criticality 1 single failure points, the items during intact aborts not meeting the fail safe redundancy requirements shall be identified in the individual element Critical Items List.

This fail safe requirement does not apply to the premature firing failure mode of pyrotechnical devices and functional systems, except associated avionics and circuity, or to the aero-surface actuators, SSME actuators, or LRB TVC * actuators when subjected to gross contamination of their hydraulic supply. The SSME and LRB shall be relieved of the fail safe operational requirements when a shutdown is prevented by vehicle applied shutdown inhibit command.

Deviation/Waivers (TBD) *

3.5.1.1.1.1 Primary Structure, Thermal Protection, Pressure Vessels. The primary structure, TPS, and pressure vessels subsystems shall be designed to preclude failure by use of adequate design safety factors, relief provisions, fracture control, or safe life and/or fail safe characteristics.

3.5.1.1.1.2 (Deleted).

3.5.1.1.2 Redundancy Verification. Redundant functional paths or subsystems shall be designed so that their operational status can be verified during ground turnaround without removal of LRUs. In addition, these redundant functional * paths of subsystems shall be designed so that their operational status can be verified inflight to the maximum extent possible, but as a minimum shall provide capability for redundancy management in the event of a malfunction of a functional path and shall provide information to the crew regarding redundancy status of the affected system sufficient to determine if a failure occurred and if an abort decision is required. Exceptions to the inflight verification requirement of redundant functional paths include:

a. Standby redundancy (redundant paths where only one path is operational at any given time)

b. All functional paths of any subsystem which is inoperative (during such inoperative periods)

c. Pyrotechnic devices

d. Mechanical linkage

Critical redundant items whose failure cannot be detected during normal ground turnaround operations or during flight shall be identified in the individual element Critical Items List. Redundancies within a functional path shall be so designed that their operational status can be verified prior to each installation into the vehicle.

Deviation/Waivers (TBD) *

3.5.1.1.3 Separation of Critical Functions. Alternate or redundant means of performing a critical function (see Paragraph 6.1.4) shall be physically separated or protected at least to the extent of separating the first means from the second means, such that an event which causes the loss of one means of performing the function will not result in the loss of alternate or redundant means. Any Deviation/Waiver to requirements for physical separation of critical functions shall be in accordance with NSTS 22206.

Deviation/Waivers (TBD) *

3.5.1.1.4 Protection of Redundant Components. Redundant components susceptible to similar contamination or environmental failure causes such as shock, vibration, acceleration or heat loads shall be physically oriented or separated to reduce the chance of multiple failure from the same causes(s).

3.5.1.1.5 Isolation of Subsystem Anomalies. Isolation of anomalies of critical functions shall be provided such that a faulty subsystem element can be deactivated either automatically or manually without disrupting or interrupting alternate or redundant functional paths or other subsystems which could cause a Criticality 1 or 2 condition. During ground operations, capability to * fault-isolate to the line replaceable unit or group of units without disconnections or use of carry-on equipment, shall be provided. This requirement shall apply to the External Tank and the Liquid Rocket Booster only when they are in the VAB or on the pad, and to the SSME only when installed in the Orbiter.

3.5.1.1.6 Arming/Disarming Explosive. Provisions shall be made for arming explosive devices as near to the time of expected use as is feasible. Provisions shall be made to promptly disarm explosive devices when no longer needed.

3.5.1.2 Ground System Reliability.

3.5.1.2.1 (Deleted).

3.5.1.2.1.1 GSE Fail Safe. All ground support equipment (except primary structure and pressure vessels) shall be designed to sustain a failure without causing loss of vehicle systems or loss of personnel capability *
GSE structure and pressure vessels shall be designed with safety factors as specified in 3.2.2.2.2.

Deviation/Waivers (TBD) *

3.5.1.2.2 GSE Failure Protection. Ground support equipment failure shall not propagate sequentially in associated support equipment or induce a failure in the flight vehicle.

3.5.1.2.3 GSE Input Verification. Ground support equipment used for flight vehicle subsystems operation, test, checkout, or maintenance shall provide for routine verification tests before a flight vehicle connection is made to assure that each fluid or electrical/electronic input to the vehicle is compatible with the related vehicle subsystems.

3.5.1.2.4 GSE Automatic Switching. Provision shall be made for automatic switching to a safe mode of operation for GSE failure modes which could result in the loss of a critical function and where there is not enough time for manual correction of the condition. Caution and warning shall be provided for these time-critical functions. *

3.5.2 Maintainability.

3.5.2.1 Shuttle Systems Maintainability. A 160-hour turnaround capability can be achieved by placing the proper attention to location and access to flight hardware during the design and development activities and through the evolution of operational techniques. The Shuttle flight systems and their LRUs shall be designed such that they are accessible and capable of being removed and installed within their allocated turnaround time. The times to remove and replace the various LRUs shall be identified and shall be demonstratable.

3.5.2.2.1 Liquid Rocket Booster/External Tank Buildup, Mating and Service. The LRB and ET shall have their respective LRUs located and access provided such that minimum time to replace or service them is achieved during the buildup, verification, and assembly of the LRB/ET. The ET and LRB shall be capable of alignment, connection, inspection, and verification of mechanical and electrical interfaces during mating operations. The capability to verify LRB/MLP, ET and Orbiter interface signatures prior to mate shall be provided. *

3.5.2.2.2 Orbiter Mating, Maintenance and Servicing. The Orbiter shall be capable of having planned maintanance performed within the time allocation specified for turnaround. The Orbiter LRUs shall be located and access provided to allow removal and replacement. The Orbiter and ET shall be capable of alignment, connection, inspection, and verification of electrical, fluid, and mechanical interfaces during the mating operations. *

3.5.2.2.3 Shuttle Flight Vehicle Checkout. The Orbiter vehicle, ET, and LRB shall be capable of checkout after ground system connection on the launch pad. Provision shall be made to allow for maintenance of appropriate LRUs in the vertical position. *

3.5.2.2.4 Shuttle Flight Vehicle Access. The Orbiter vehicle, ET, and LRB shall be capable of access to equipment installations, element interfaces, and service umbilicals requiring inspection, servicing, installation, or verifica- tion. *

3.5.2.2.5 Shuttle Element Turnaround Allocations. See NSTS 07700, Volume IX for the Timeline Allocations. *

3.5.2.3 Ground Systems Maintainability. The operational ground system including equipment and facilities, shall provide turnaround support of flight hardware during Shuttle ground operations within the specified turnaround allocation defined in Paragraph 3.5.2.1. The design of the ground support equipment and facilities shall consider fault isolation, ease of replacement of failed components, and operation manpower as part of the design consideration.

3.5.2.3.1 Turnaround Support. The ground system shall be capable of a minimum service life in accordance with Paragraph 3.5.3.1.

3.5.2.3.2 Turnaround Flow. Ground system maintenance, refurbishment and revalidation of turnaround facilities of in-line refurbishment of turnaround support equipment shall not interfere with flight vehicle operations. The support equipment hardware shall be packaged for ease of access and replacement of component parts.

3.5.2.3.3 Shuttle Flight Vehicle Access. The mobile launch platform, service tower, VAB and pad shall be capable of providing access to flight vehicle interfaces, equipment installations, and service umbilicals requiring inspection, servicing, installation, or verification.

3.5.2.3.4 Launch Preparation. The crawler/transporter, mobile launch platform, service tower, and pad shall be capable of supporting flight vehicle launch preparation and launch activities in a timeframe compatible with the traffic model.

3.5.3 <u>Useful Life</u>.

3.5.3.1 Reuse. The Shuttle System operational capability shall be maintained at the flight rate specified in NSTS 07700, Volume III (FDRD). Each NSTS element shall maintain adequate visibility through compliance with the NSTS System Ingetrity Assurance Program Plan (NSTS 07700, Volume XI) to predict loss of the capability 5 years in advance.

3.5.3.1.1 Orbiter Vehicle. Each Orbiter vehicle shall be mission capable for a minimum of 10 years from delivery to NASA, with a documented design-life extension certification program that defines useful life. Each Orbiter vehicle shall be capable of performing 100 orbital missions, including ground turnaround operations, with scheduled subsystem maintenance and/or refurbishment, or replacement, as determined to be most cost effective. The TPS for the Orbiter vehicle shall be capable of performing 100 orbital missions, with unscheduled subsystem maintenance and/or refurbishment, or replacement, as determined to be most cost effective. TPS thermal/structural design shall be based on a 100 mission life. Structural design shall be based on a least 100 mission life. As a goal, the Orbiter vehicle structure should be capable of 500 reuses. The actual reuse capability will be determined as the flight program progresses through analysis of flight results and/or additional structural tests. Refurbishment and/or modifications to achieve this capability are premissible.

3.5.3.1.2 External Tank. The ET shall be capable of launch within 5 years of delivery to NASA.

3.5.3.1.3 Liquid Rocket Booster. (TBD) *

3.5.3.1.4 Carrier Aircraft. The carrier aircraft shall have an operating life of 270 ferry missions.

3.5.3.1.5 Main Engine. The SSME shall be capable of a service life, between overhaul, of 55 starts or 7 1/2 hours (27,000 seconds) total time including not more than 14,000 seconds at FPL.

3.5.3.1.6 Liquid Rocket Booster Engine - Service Life. (TBD) *

3.5.4 <u>Safety</u>. Shuttle System safety shall be in accordance with NHB 5300.4(1D-2).

3.5.4.1 Flight System Safety.

3.5.4.1.1 Safety Design Preferences. The flight vehicle shall, in the following order of preference, be designed to eliminate hazards by * appropriate design measures; or prevent hazards through use of safety devices or features; or control hazards through use of warning devices, special procedures, and emergency protection subsystems.

3.5.4.1.2 Crew Warning and Emergency Provisions. The flight vehicle shall have capability to provide crew warning of hazardous conditions and provisions for corrective action, emergency crew and passenger egress/escape, abort action, or mission termination.

3.5.4.1.3 (Deleted).

3.5.4.1.4 Materials. Flight vehicle materials shall be selected with characteristics which do not present hazards to personnel or equipment in their intended use or environment.

3.5.4.1.5 Isolation of Hazardous Conditions. Provisions shall be made to physically isolate or separate hazardous, incompatible subsystems, materials, or environments. Designs shall consider space flight hazardous conditions identified in MSC-00134.

3.5.4.1.6 Purging, Venting, Drainage, Detection. Provisions shall be made to prevent hazardous accumulations of gases or liquids in the flight vehicle (i.e., toxic, explosive, flammable or corrosive). Detection of hazardous gases shall be required in critical areas and closed compartments during ground operations, even where ground supplied purge is provided, to insure no hazardous conditions exist. A redundant/alternate hazardous gas detection capability shall be required to prevent a launch delay or a launch scrub, if the primary hazardous gas detection system is lost.

3.5.4.1.7 Drain, Vent and Exhaust Port Design. Flight vehicle drains, vents, and exhaust ports shall prevent exhaust fluids, gases, or flames from creating hazards to personnel, vehicle, or equipment.

3.5.4.1.8 Protection of Critical Functions. Flight vehicle subsystems shall be designed to prevent inadvertent or accidental activation or deactivation of safety-critical functions or equipment, which would be hazardous to personnel or vehicles during flight and ground operations.

3.5.4.1.9 Battery Protection. Flight vehicle batteries shall be isolated and/or provided with safety venting systems and/or explosion protection.

3.5.4.1.10 Flight Vehicle Separation. Flight vehicle subsystems or equipment which are severed or disconnected during mission events (e.g., staging) shall not degrade mission success or crew safety.

3.5.4.1.11 Pressure Vessel Protection. Pressure vessels shall be protected against overpressurization or underpressurization which could be hazardous to personnel or flight vehicle.

3.5.4.1.12 Range Safety. The Flight Termination System shall comply with the range safety Flight Termination System requirements of ESMCR 127-1 and WSMCR 127-1. The flight vehicle shall comply with the range safety requirements of WSMCR 127-1. In those instances where adherence is judged to be inappropriate from either an operational or technical standpoint, such instances shall be brought to the attention of the DOD/NASA for resolution. The design, performance, development, acceptance, and qualification test requirements for a Shuttle Range Safety Command Destruct System to be used on the Liquid Rocket Booster (LRB), Left and Right, and the External Tank (ET) of the Space Shuttle Vehicle shall be in accordance with MSFC-SPEC-30A90506 Shuttle Range Safety Command Destruct System, Specification for. *

3.5.4.1.13 Flammable Gas Concentration Limit. The flight vehicle shall be designed to preclude the concentration of flammable gases in critical areas and closed compartments from exceeding the lower flammable limit for the combination of gases that may be present in areas or compartments for prelaunch, flight, and postlanding operations. Specific prelaunch redlines shall be established to ensure hazardous concentrations are not exceeded in flight. Provide the capability to have a hazardous gas detection system that is capable of periodic sampling of concentrations of hydrogen, hydrocarbons, and oxygen. *

3.5.4.2 Ground System Safety.

3.5.4.2.1 Ground Support Equipment and Facilities. GSE and facilities shall be designed to preclude and/or counteract failures or hazards that would jeopardize personnel safety or damage or degrade the vehicle, GSE, and facilities.

3.5.4.2.2 Flight Vehicle Safing for Ground Operations. Flight vehicle safing shall be provided by GSE during ground turnaround, maintenance, and refurbishment operations.

3.5.4.2.3 Emergency Egress. The ground system shall facilitate emergency egress of flight crews, passengers, and ground crews to a safe area during all ground operational phases.

3.5.4.2.4 Hazardous Gas Detection and Disposal. The ground system shall provide for safe disposal of hazardous vented or boil-off gases. Detection of hazardous gases shall be required in ground systems critical areas and closed compartments where such detection is critical to personnel safety or ground operations.

3.5.4.2.5 Flight Vehicle Handling and Safing. The ground system shall provide protection of personnel and equipment during safing and handling a vehicle following return from a mission.

3.5.4.2.6 Air Liquefaction. Shuttle Systems shall be designed to prevent air liquefaction external to cryogenic systems which could present a hazard to personnel, hardware, or cause operational anomalies during ground operations.

3.5.4.2.7 Range Safety. The flight vehicle and ground support equipment shall comply with the Range Safety requirements of WSMCR 127-1. In those instances where adherence is judged to be inappropriate from either an operational or technical standpoint, such instances shall be brought to the attention of the DOD/NASA for resolution. *

3.5.4.2.8 Material Handling Equipment and Operating Personnel. Material handling equipment and operating personnel shall be certified in accordance with the requirement of NSTS 08114, Shuttle Program Requirements for Periodic Certification of Material Handling Equipment and Operating Personnel.

3.5.5 Human Performance.

3.5.5.1 Personnel Skill Requirements. The flight vehicle shall not require personnel skills more demanding than those required for operational, high performance land-based aircraft systems.

3.5.5.2 Sizing of Personnel. The flight vehicle shall provide furnishings, equipment, work spaces, and access-ways sized for the following design populations:

a. Accommodations used exclusively by the commander and pilot shall be sized for personnel within the 5th to 95th percentile dimensions of the dimensions of the USAF male population as extrapolated to 1980. However, the accommodations shall be readily adaptable to accommodate an individual as small as a 5th percentile female as identified in USAF AMRL-TR-70-5 (1968 USAF Women).

b. All other accommodations shall be sized for personnel within the dimensional range of the 5th percentile female (based upon the 1968 USAF women) to the 95th percentile male (based upon the extrapolated 1980 USAF male).

c. The designs of custom tailored equipment, such as flight clothing and EMUs shall be capable of accommodating, on an as required basis, any individual from a 5th percentile female (based upon the 1968 USAF women) to the 95th percentile male (based upon the extrapolated 1980 USAF male).

3.5.5.3 Human Engineering Criteria. MIL-STD-1472 shall be used as a guide for human engineering design criteria. The touch temperature of crew related * flight equipment shall comply with NSTS 07700, Volume X, Paragraph 3.3.1.2.4.2, Crew Exposure (maximum temperature).

3.5.6 Transportability. Each Shuttle System element (element components), when protected in accordance with the requirements of (TBD) * shall be capable of being handled and transported from its fabrication site, to its final operational or launch position, without degradation of reliability. The condition of flight elements after transport shall be acceptable, subject to wear from normal use during transport modes.

3.5.6.1 Handling, Packaging and Transportation Compatibility. Shuttle System elements (or element components) shall be compatible with the handling, packaging, and transportation systems to the extent that:

a. The size and weight of the element or element component does not exceed the limitation of feasible handling, packaging, and transportation systems

b. No loads are induced in the element during transportation and handling which will produce stresses, internal loads, or deflections in excess of that for which the element has been designed and certified; and

c. The element is adequately protected against natural environments during transportation and handling

3.5.7 Hazardous Materials and Components. Hazardous materials and components (i.e., fuels, oxidizers, pyrotechnic devices) shall be used, handled and maintained in a manner that will not constitute a hazard to personnel, vehicle, equipment, payloads and/or the mission.

3.6 SYSTEM DESIGN AND CONSTRUCTION STANDARDS.

3.6.1 Selection of Specifications and Standards. Specifications and standards for use in the design and construction of the Space Shuttle System shall be selected in accordance with MIL-STD-143, except that NASA documents, where specified shall take precedence.

3.6.1.1 Commonality. The design of the Space Shuttle System shall provide for maximum efficiency of equipment selection and/or development through multiple applications of common items. Common items and their applications shall be identified, selected and implemented in accordance with the commonality requirements of NSTS 07700, Volume VII, Commonality Management. *

3.6.2 Materials, Parts and Processes.

3.6.2.1 Materials and Processes. Materials and processes, except those for new GSE, shall be selected in accordance with JSC-SE-R-0006. GSE covered by JSC SE-R-0006 shall be limited to only that equipment which enters the vehicle or to equipment where GSE hazardous fluid/gas materials compatibility or induced contaminations can adversely affect flight hardware. The design requirements for new GSE are defined in Paragraph 3.6.16 of this document.

3.6.3 Parts Selection. EEE, mechanical, and fluid parts shall be selected from the applicable element project parts list.

3.6.3.1 Electrical Connector Restriction. Electrical connector configurations as described in MSFC-SPEC-40M38298, used to connect pyrotechnic firing circuits to the NASA standard initiator, type 1 (NSI-1) are restricted to the specific configurations listed below:

a. NBS8GE8-2SE Connector Configuration shall be used to connect ET/LRB strut pyrotechnic firing circuits to the NSI-1. Note - when using this connector, assure that there is adequate structural clearance. *

b. NBS9GE8-2SE, -2F, AND -2SH Connector Configurations shall be used to connect all other pyrotechnic firing circuits to the NSI-1.

3.6.4 Moisture and Fungus Resistance. Materials which are non-nutrient to the fungi defined in MIL-STD-810, Method 508 should be used. When fungus nutrient materials must be used, they should be hermetically sealed or treated to prevent fungus growth for the effective lifetime of the component. Materials not meeting this requirement shall be identified as a limited life component and shall identify any action required such as inspection, maintenance, or replacement periods. Fungus treatment should not adversely affect unit performance or service life. Materials so treated should be protected from moisture or protective agent. Fungus inert materials are listed in MIL-STD-454 (Requirement No. 4).

3.6.5 Corrosion of Metal Parts.

3.6.5.1 Flight System Corrosion.

3.6.5.1.1 Stress Corrosion. JSC SE-R-006 shall be used for design and materials selection for controlling stress corrosion cracking, except MSFC-SPEC-522 shall be used in place of MSFC-DWG-10M33107 as required in the second sentence of Paragraph 3.6.4. The requirement to use MSFC-SPEC-522A is effective as of August 1, 1978, for new design and need not be applied retroactively.

3.6.5.1.2 Corrosion Protection. Corrosion resistant metals shall be used wherever possible. The use of dissimilar metals, finishes, and coatings shall comply with the requirements of MSFC-SPEC-250.

3.6.5.2 Ground System Corrosion.

3.6.5.2.1 Corrosion Protection. The use of dissimilar metals, finishes, and coatings shall comply with the requirements of MSFC-SPEC-250.

3.6.6 Interchangeability and Replaceability. The definitions of item levels, item exchangeability, models and related items, shall be in accordance with MIL-STD-280.

3.6.6.1 Flight Vehicle Interchangeability. The flight vehicle interfaces shall allow interchangeability between any production liquid rocket booster, external tank and Orbiter vehicle, or between any production Orbiter vehicle or payload module that may be selected to be mated or installed. Interchangeability of selected major subassemblies shall be possible, e.g. OMS, RCS and APU modules; landing gear; and hydraulic actuators. *

3.6.6.2 Replaceability of Hardware. The Shuttle Flight Vehicle hardware shall be interchangeable except for those selected items which will be replaceable. *

3.6.7 Electromagnetic Compatibility. The Shuttle System and elements thereof including payloads, shall be designed and tested in accordance with NSTS SL-E-0001, Specification, Electromagnetic Compatibility Requirements, Systems for the Space Shuttle Program. Subsystem and/or individual equipment shall be designed and tested in accordance with the following documents:

a. NSTS SL-E-0002, Specification Electromagnetic Interference Characteristics, Requirements for Equipment, for the Space Shuttle Program.

b. MIL-STD 462, Electromagnetic Interference Characteristics, Measurement of.

c. MIL-STD-463, Definition and System of Units, Electromagnetic Interference Technology.

The subsystem and/or individual equipment requirements are not applicable to ground system procurements unless specifically required by the procuring activity to meet the requirements for EMI critical equipment as defined in NSTS SL-E-0001.

3.6.8 Identification and Marking. The identification and marking of Shuttle System equipment shall be in accordance with MIL-STD-130, except that the "design activity code", manufacturer's trademark" and "licensor code identification", need not be combined with the part number when marking parts and assemblies. The identification and marking of GFE furnished by JSC may be in accordance with MIL-STD-130 or MSC-SPEC-M-1. Pipe, hose and tube lines of flight vehicles only shall be marked in accordance with MIL-STD-1247. Ground Support Equipment fluid lines and compressed gas cylinders shall be marked in accordance with MIL-STD-101. Existing GSE/facility piping installed at KSC Launch Complex 36 shall remain as currently identified; this equipment has been identified in accordance with MIL-STD-1247 and shall be treated as a unique case within the National STS Program. New GSE/facility piping required to interface with existing Launch Complex 36 equipment shall be identified in accordance with Space Transportation System Payload Ground Safety Handbook KHB 1700.7 and MIL-STD-101B. Direct electro-chemical etched markings may be used when other marking is not feasible. Packing marking requirements shall conform to the requirements of MIL-STD-129.

3.6.8.1 (Deleted).

3.6.8.2 Interface Identification. All interface fluid, gaseous, mechanical and electrical connections (element-to-element, element-to-payload, ground-to-flight) will be identified in a manner to provide ease of viewing, with and without GSE installed, with the flight element in either horizontal or vertical position.

3.6.8.3 Element Cosmetic Coatings. Cosmetic requirements for all Shuttle elements shall be restricted to appropriate markings or decals, as necessary. Priority consideration shall be given to weight and thermal performance.

3.6.9 Storage. The Shuttle System hardware shall be designed for a storage life in accordance with the storage requirments defined in the respective element end item specifications, except that in those cases where age-sensitive materials cannot be avoided, replacement of such materials shall be permitted on a scheduled basis during the storage period.

3.6.10 Drawing Standards. Refer to NSTS 07700, Volume IV, Appendix E.

3.6.11 Coordinate System Standards. Coordinate system standards for the Shuttle System are defined in JSC 09084.

3.6.12 Contamination Control.

3.6.12.1 System Contamination Control. Contamination of the Space Shuttle System shall be controlled to assure system safety, performance, and reliability. Control shall be implemented by a coordinated program from design concept through procurement, fabrication, assembly, test, storage, delivery, operations, and maintenance of the Shuttle System. This program shall comply wiith the requirements of SN-C-0005, Specification Contamination Control Requirements for the Space Shuttle Program. Selection of system design shall include self-cleaning (filtering) protection compatible with component sensitivity.

Wire cloth filters used in the flight vehicle shall conform to NSTS Specification SE-F-0044.

Specific cleanliness levels shall be established for material surfaces, fluid systems, functional items, and habitable areas as required for effective control of contamination.

Fluid particulate cleanliness shall be maintained at acceptable levels for fluids used to service flight elements or major test articles by the use of either a qualified interface filter, a qualified final filter, or approved final filter rationale as specified in SE-S-0073.

Final filters, interface filters, and interface filter/disconnect assemblies shall be qualified as specified in SE-S-0073.

Fluids used in acceptance and qualification of components, and subsequently in assembly of or use in higher level assemblies, subsystems, or systems for verification and operation shall meet the purity cleanliness, and analysis requirements of NSTS SE-S-0073.

NSTS 08131, Space Shuttle System Contamination Control Plan, documents the overall program contamination control tasks and responsibilities.

Equipment designed specifically for the Space Shuttle program shall comply with the specified requirements. Selection of off-the-shelf equipment for application to the Space Shuttle program shall comply with the intent of these requirements.

3.6.12.2 Operational Contamination Control. Contamination Control during the operational phases of the Space Shuttle is necessary to insure overall satisfactory performance of the system. Of particular concern is the gaseous and particulate environment of the Orbiter during all operational phases. Because of the wide range of payloads it is the objective of the following approach to provide requirements to satisfy the needs of the large majority of payloads. Payloads that have special requirements not covered herein shall provide the necessary system(s) to satisfy such requirements. Although operational phase of the system will be covered primarily, specific requirements which affect design of the elements of the system are included.

The following requirements will be incorporated in the generation of the contamination control plan required in Paragraph 3.6.12.1.

3.6.12.2.1 Element Cross Contamination. Space Shuttle System element design and operation shall be such as to minimize cross contamination of the elements to a level compatible with mission objectives.

3.6.12.2.2 Payload Bay Design. The payload bay shall be designed to minimize contamination of payload and critical payload bay surfaces to a level compatible with mission objectives. Orbiter elements which are not easily cleaned, e.g., internal ribbed structure, OMS kits, door actuators, etc., and elements which are sources of particulate, vapor, VCM (Volatile Condensible Material), or other contamination, shall be isolated from the payload and critical payload bay surfaces. All nonmetallic materials exposed to the payload shall be selected for outgassing characteristics as specified in Paragraph 3.6.2.1. The payload bay shall be designed to protect critical payload and payload bay surfaces from contamination by the external environment during any closed payload bay door operational phase of the Space Shuttle System.

3.6.12.2.3 Payload Design. Critical surfaces such as Orbiter radiators, windows, optics, etc., within the payload bay and part of the Orbiter System must be protected in the same manner as payloads. That is, payloads must insure that their effluents and operations do not jeopardize the performance of these systems. Payloads shall comply with the requirements of Paragraph 3.6.2.1 and also shall provide cleanable exterior surface.

3.6.12.2.4 Operational Capabilities. The Space Shuttle System shall provide the capability for satisfying the following requirements.

3.6.12.2.4.1 Payload Loading and Checkout. Prior to payload loading the internal surfaces of the payload bay envelope shall be cleaned to a visibly clean level, as defined in SN-C-0005. This cleaning shall be accomplished within a protective enclosure in order to isolate sources of contamination from critical regions. This enclosure shall be continuously purged with nominally class 100, guaranteed class 5000 (HEPA filtered) air per FED-STD-209 and shall contain less than 15 parts per million hydrocarbons, based on methane equivalent. The air within the enclosure shall be maintained at 70 $\pm$ 5^{o}F and 50% or less relative humidity. The payload loading operation shall be accomplished so as to avoid contaminating the payload and payload bay by temperature, humidity, and particulates consistent with requirements specified herein. More stringent particulate and relative humidity requirements may be implemented on particular payloads pending technical justification of the requirement.

3.6.12.2.4.2 Contamination Control Subsequent to Payload Loading. Subsequent to payload loading, accumulation of visible particulate and film contamination on all surfaces within the payload bay shall be prevented by controlled work discipline and cleanliness inspections and effective cleaning as required. The air purge, temperature, and humidity requirements of the above Paragraph 3.6.12.2.4.1 shall be maintained.

3.6.12.2.4.3 Preparation for Closeup of Payload Bay. Prior to final closure of the payload bay in preparation for vehicle mating, inspection and cleaning, as required shall be conducted to verify that all accessible surfaces within the payload bay, including external surfaces of payloads, meet the visibly clean level stipulated in the above Paragraph 3.6.12.2.4.1. When payload changeout in the vertical configuration is required, the purge gas class, temperature, and humidity requirements of the above Paragraph 3.6.12.2.4.1 shall apply.

3.6.12.2.4.4 Closed Payload Bay Operations. The Orbiter shall be designed for closed payload bay purging by GSE, subsequent to payload bay closure using conditioned purge gas (air or GN_2) which has been HEPA filtered, class 5000, and contains 15 ppm or less hydrocarbons based on methane equivalent. Continuous purging will be supplied except during switchover between mobile and facility GSE at the OPF, VAB, PAD, and during towing from the OPF until Orbiter mating operations are complete in the VAB.

3.6.12.2.4.5 Launch Through Orbit Insertion.

3.6.12.2.4.5.1 Cleanliness Levels. The level of cleanliness maintained at preflight on the payload and payload bay, shall be retained through launch to orbital insertion including lift-off, LRB separation, etc. *

3.6.12.2.4.5.2 Purging. Any purging, other than that provided by normal depressurization of the payload bay or payloads during this operational phase, shall be the responsibility of the payloads.

3.6.12.2.4.6 On-Orbit. Overboard venting of gases or liquids shall be controlled either in design or operation to avoid contamination of the payloads, payload bay, Orbiter windows, optical surfaces, or Orbiter thermal protection system surfaces to a level compatible with mission objectives. Food, water, and waste vents shall be as defined in Paragraph 3.3.1.2.4.5. As a design and operational goal, venting of gases and liquids from the Orbiter will be limited for sensitive payloads to control in an instrument field of view particles of 5 microns in size to one event per orbit, to control induced water vapor column density to 10^{12} molecules/cm^2, or less, to control return flux to 10^{12} molecules/cm^2/sec., to control continuous emissions or scattering to not exceed 20th magnitude/sec^2 in the UV range, and to control to 1% the absorption of UV, visible, and IR radiation by condensibles on optical surfaces. Materials which can contaminate either the payload, payload bay or Orbiter windows by outgassing when exposed to the vacuum environment shall be selected for low outgassing characteristics as defined in Paragraph 3.6.2.1.

RCS thruster firing operations shall be planned to avoid contamination particularly when the payload bay doors are open. Thruster exhausts shall be designed and controlled in operation to minimize direct impingement or reflection upon the deployed or released (attached or unattached) payload or open payload bay. RCS engine design and operation shall consider the minimization of contamination. The design of other devices to be operated in flight, such as the mechanical manipulator, shall be such that the generation of contamination is controlled to a level compatible with mission objectives.

3.6.12.2.4.7 Reentry Phase (Deorbit to GSE Attachment). The payload bay shall be repressurized using filtered atmospheric air (50 microns absolute). No control of humidity or concentrations of other gases will be provided by the Orbiter.

3.6.12.2.4.8 Post-Landing.

3.6.12.2.4.8.1 Primary Landing Station. The Orbiter design and related GSE shall include the capability for closed payload bay purging subsequent to landing as defined in Paragraphs 3.2.1.1.15 and 3.6.12.2.4.4. The payloads will be removed in the environment as defined in Paragraph 3.6.12.2.4.1, if required.

3.6.12.2.4.8.2 Secondary Landing Station. No special requirements.

3.6.12.2.4.8.3 Contingency Landing Station. No special requirements.

3.6.13 <u>Traceability</u>. Traceability shall be provided by assigning a traceability identification to each system element identified in Paragraph 3.1.1.1 and providing a means of correlating each to its historical records, and conversely, the records must be traceable to each system element. Ground operations system traceability requirements shall be in accordance with the requirements of Paragraph 3.4.15 of Space Shuttle Ground Support Equipment General Design Requirements, SW-E-0002.

3.6.13.1 Traceability Classification. Traceability classification is the classification of a raw material, part, assembly, or end item for determining the traceability marking and traceability records required or excluded for the item. Engineering Documentation (e.g., specifications and drawings) shall specify traceability for items in accordance with the following:

a. <u>Serial Traceability (TS)</u> - Hardware assemblies and components down to and including the Line Replaceable Unit (LRU) level, shall be traceable by serial where one or more of the following apply:

    1. The item is contained in the Critical Items List (CIL)

    2. The item has a limited useful life

    3. The item is to be subjected to acceptance induced environmental test (thermal and/or vibration)

    4. The item requires progressive comparative measurements of performance (i.e., transducer curves)

    5. The item is subject to fracture control

    6. The item contains traceable subordinate units, assemblies, or parts

b. <u>Lot Traceability (TL)</u> - This classification requires lot serial numbering on items produced (manufactured, processed, inspected, or tested by the batch, mix, heat, or melt) in given time sequence, without changes in materials (substitutions); changes in tooling or processes (which would affect form, fit or function);
or substitution of non-certified personnel for those normally requiring certification; and without change in configuration. The "given time sequence" nominally includes identification of work from the initiation of the production order for specific hardware manufacture, through completion of the last operation on the production order, and therefore includes accumulation of generic data which are related to all items of a particular lot. Electrical, Electronic, and Electromechanical (EEE) parts specified in "applicable element project parts list," require lot traceability as a minimum.

c. Member Traceability (TM) - Both serial number and lot number traceability shall be required on items which must be identified in such a manner that they can be handled as members of a lot and also controlled as individual items.

d. Exempt from Traceability (E) - All items not falling into one of the previous classifications shall be classified as exempt.

3.6.13.2 Traceability Identification. Each item identified as traceable (TS, TL, TM) shall have a traceability identifier consisting of the manufacturer's DOD code identification number and a serial, lot, or member number. The serial, lot, or member number shall be assigned by the manufacturer and shall not exceed ten characters (alphas, numerics, dashes, etc.).

3.6.14 Electrical Bonding. Electrical bonding shall be in accordance with MIL-B-5087 in all areas, except in the area of lightning protection where the requirements of NSTS 07636 shall apply.

3.6.15 Electrical Installations.

3.6.15.1 Soldering. Soldering of electrical connectors shall be in accordance with NHB 5300.4 (3A), as supplemented by JSC 08800.

3.6.15.2 Circuit Boards. Single and double sided printed wiring board assemblies shall be designed, documented, and fabricated in accordance with MSFC-STD-154. Multilayer printed wiring board assemblies shall be designed and documented in accordance with NSTS Specification SN-P-0006. The fabrication of multilayer printed wiring board assemblies for flight hardware only shall be controlled by NSTS Specification SN-P-0006. Parts mounting design requirements for all types of printer wiring board assemblies shall be in accordance with MSFC-STD-136. GSE is excluded from this requirement.

3.6.15.3 Moisture and Fungus Resistant Treatment. Electrical, electronic and communications equipment shall be treated for moisture and fungus in accordance with requirements specified in Paragraph 3.6.4.

3.6.16 GSE/Facility Design. New facilities to be utilized at KSC shall be designed in accordance with NHB 7320.1. New ground support equipment to be utilized in the Space Shuttle Program shall be designed in accordance with NSTS SW-E-0002. GSE not classified as new design shall meet the Materials and Processes requirements specified in Paragraph 3.6.2.1 of this document.

3.6.17 Screw Threads. Screw threads for threaded fasteners used on Shuttle System hardware shall be of unified thread form in accordance with MIL-S-7742 or MIL-S-8879, as applicable:

a. Material strength levels up to, but not including 160 KSI may be threaded per MIL-S-7742 or MIL-S-8879. Rolled threads are preferred.

b. Material strength levels 160 KSI and above shall be threaded per MIL-S-8879. Any rolling of external threads, when required, shall be done after heat treatment.

c. Proprietary design blind bolt screw threads used on the ET shall be subject to requirements of their procurement specifications.

d. The ET external threads of threaded inserts shall not be subject to the requirements of these specifications. These threads may have a Class 2 tolerance range and have a modified minor diameter.

e. The ET internal thread forms and tolerances of the shock isolator and mounting fasteners for DFI electronic equipment mounting provisions are not subject to this requirement.

Screw threads used on airborne fluid systems fitting shall be of unified thread form, Class 3 in accordance with MIL-S-7742 or MIL-S-8879. The body and jam nut of the ET electrical feedthru connector may have Class 2 threads.

3.6.18 Contract End Item (CEI) Specification Format. The CEI specifications for the Shuttle System elements shall be prepared in accordance with NSTS 07700, Volume IV.

The CEI Specification for Shuttle computer systems and software shall be prepared in accordance with NSTS 07700, Volume XVIII.

3.6.19 Design Criteria and Standards. Shuttle System Flight and Ground Systems shall conform to the individual standards of JSCM 8080 identified in Table 2.0 of the basic Volume X. Each element project office will provide a plan describing the method and extent of implementation of each of the standards to the SSPO for information. Whenever equivalent standards exist at other NASA Centers the element project office may specify these other standards as an alternative and Table 2.0 will be revised to reflect this substitution. Policy relative to the method and extent of implementation of each of the standards towards GFE used with the Shuttle systems is provided by JSC Management Instruction 8080.2.

3.6.20 Shuttle System Pyrotechnics. All pyrotechnics and associated electrical circuits and electronics shall conform to the Space Shuttle System Pyrotechnic Specification, NSTS 08060.

3.6.21 Lightning Protection. The Shuttle System and elements thereof shall be designed and tested in accordance with NSTS 07636. NSTS 20007 is to be used for verification that the vehicle design meets the requirements criteria document NSTS 07636, and specifically identifies the analysis and test methods to be used for new and existing equipment.

3.6.22 Seismic Protection. All GSE used in close proximity to Space Shuttle Vehicle (SSV) elements, or GSE that can otherwise cause damage to SSV elements by virtue of their operation, or failure to operate during a seismic event, shall be designed considering the hazards defined in Section XVI of TM-82473, and in accordance with SW-E-0002, Appendix A.

3.7 QUALITY ASSURANCE. Shuttle System quality shall be in accordance with NHB 5300.4(1D-2).

3.7.1 Inspection Requirements. Nondestructive inspection requirements for materials and parts shall be in accordance with MIL-I-6870.

3.7.2 Sampling Requirements. Sampling requirements shall be in accordance with MIL-STD-105 and MIL-STD-414.

3.7.3 He and N2 Leakage Measurement. Leakage measurement of helium and nitrogen test gases shall be in accordance with MSC SE-G-0020.